CONFIDENTIAL

Copy No. 7

Issue Date: 18 April 1962

(Unclassified Title)
DESIGN STUDY OF A LARGE UNCONVENTIONAL LIQUID PROPELLANT ROCKET ENGINE AND VEHICLE

Prepared by

AEROJET-GENERAL CORPORATION
Liquid Rocket Plant
Sacramento 9, California

Final Report
Report No. LRP 257

Volume 4: Appendixes

Contract NAS 5-1025

Prepared for

OFFICE OF LIQUID ROCKETS
NASA HEADQUARTERS
Code MLPL (Mr. H. Burlage)
400 Maryland Avenue, S. W.
Washington 25, D. C.

DECLASSIFIED NASA EQUITY
By: Kelly Edwards
Printed Name/Office: AD50/MSFC
Date: 9-20-2004

EXCLUDED FROM AUTOMATIC REGRADING; DOD DIR 5200.10 DOES NOT APPLY

THIS DOCUMENT CONTAINS INFORMATION AFFECTING THE NATIONAL DEFENSE OF THE UNITED STATES WITHIN THE MEANING OF THE ESPIONAGE LAWS, TITLE 18, U. S. C., SECTIONS 793 AND 794. THE TRANSMISSION OR THE REVELATION OF ITS CONTENTS IN ANY MANNER TO AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW.

Aerojet-General® CORPORATION
LIQUID ROCKET PLANT • SACRAMENTO, CALIFORNIA
THE GENERAL TIRE

CONFIDENTIAL

TABLE OF CONTENTS

APPENDIX

A	HEAT TRANSFER
B	COMPARISON OF COST CRITERIA
C	RECOVERY POTENTIAL FOR HIGH-VELOCITY BOOSTERS
D	HIGH-THRUST-PER-INJECTOR ELEMENT
E	RELIABILITY COMPARISONS
F	TRAJECTORY AND VEHICLE ASSUMPTIONS
G	STEADY-STATE ANALYSIS OF A SIX-MILLION-POUND THRUST LIQUID OXYGEN/LIQUID HYDROGEN BOOSTER ENGINE
H	STUDY OF A LARGE PRESSURE-FED BOOSTER STAGE
I	PROPELLANT PERFORMANCE AND PROPERTIES

APPENDIX A

HEAT TRANSFER

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	1
II.	REGENERATIVE COOLING	2
III.	CONVECTIVE FILM COOLING	6
IV.	UNCOOLED NOZZLES	15

FIGURE LIST

	FIGURE NO.
Regeneratively Cooled 2,000K DeLaval Nozzle	1
Regeneratively Cooled 2,000K DeLaval Nozzle, Gas-Side Steel Temperatures	2
Regeneratively Cooled 2,000K DeLaval Nozzle, Gas-Side Heat Transfer Coefficients	3
Heat Exchanger Design Requirements for Heated-Hydrogen Cycle, 2,000K Forced-Deflection Engine	4
Heated-Hydrogen Cycle, 2,000K Forced-Deflection Nozzle--Identification of Heat Exchanger Stations	5
Heated-Hydrogen Cycle, 2,000K Forced-Deflection Engine--Gas-Side Steel Temperatures	6
Heated Hydrogen Cycle, 2,000K Forced-Deflection Engine--Heated-Hydrogen Temperature	7
Heated-Hydrogen Cycle, 2,000K Forced-Deflection Engine--Gas-Side Heat Transfer Characteristics	8
Required Gaseous Film Coolant vs Ratio of Gas to Coolant Velocities	9
2,000K DeLaval Nozzle, Radiation-Cooled--Equilibrium Wall Temperatures	10

I. INTRODUCTION

Heat transfer analyses of combustion chamber and exit nozzles of concepts under investigation for the 2,000,000 lb, 1,000 psi LOX/LH_2 booster were performed.

Methods of cooling the conventional (deLaval) nozzle which have been considered are:

A. Regenerative cooling with fuel

B. Regenerative cooling with a combination of fuel and turbine exhaust gases

C. Convective film cooling

Uncooled nozzles which encompass so-called radiation cooling, ablative cooling, and composite walls were also considred.

The forced-deflection nozzle which analyzed specifically for the heated hydrogen cycle, wherein hydrogen at 1,000°R is necessary to operate the engine's turbines. This, in effect, is also a regenerative cooling process, except the restriction that the coolant must be heated is added.

Conventional methods for cooling could be applied to this type of nozzle as well as to the deLaval nozzle. In this discussion, types of cooling and the requirements for the nozzle considered are presented. It is not the intention of this format to convey the impression that any method of cooling is peculiar to the type of nozzle discussed.

II. REGENERATIVE COOLING

Forced convection by coolant circulating through tubular-walled combustion chambers and nozzle exit sections is a proven method for keeping combustion chambers intact. In all instances fuel was selected as the coolant for large boosters. Therefore, in addition to maintaining the wall at a given temperature limit, the allowable pressure drop in the heat exchanger was limited to the difference between pump outlet pressure and injector face pressure.

For reasons of economy, all designs were limited to steel tubing with maximum allowable wall temperatures of 1,500°F. In some instances, an insulation-type coating was found to be necessary, and Rokide Z was selected as the most practical; however, any proved substance with equivalent insulation thickness (x/k ratio) could be substituted.

Regenerative cooling heat transfer analyses were performed by the digital computing facility of the Aerojet-General Liquid Rocket plant by means of its General Heat Transfer Program. It has been developed expressly for the purpose of analyzing regeneratively cooled chambers.

Combustion chamber or gas-side heat transfer coefficients are evaluated by the expression,

$$h_g = 0.027(k/D)Re^{0.8}Pr^{0.4} \qquad \text{(Eq 1)}$$

where D is chamber diameter, and k, Re, and Pr are thermal conductivity, Reynolds Number and Prandtl Number, respectively, of the combustion exhaust products.

Transport properties of combustion gases are evaluated at the arithmetic mean boundary temperature, $T_{am} = 1/2(T_o + T_{wg})$.

Coolant-side heat transfer coefficients are computed by the expression

$$h_c = 0.0217(k/D_e)Re^{0.8}\,Pr^{0.4}\,(T_B/T_{WL})^{0.34} \qquad \text{(Eq 2)}$$

where D_e is equivalent tube diameter, k, Re, and Pr, are thermal conductivity, Reynolds number, and Prandtl number of hydrogen, respectively. T_B is bulk coolant temperature and T_{WL} is coolant-side wall temperature. In this expression, coolant properties are evaluated at the bulk temperature, and the quantity $(T_B/T_{WL})^{0.34}$ is an adjustment factor.

Transport properties of the combustion products were evaluated by a subroutine in the digital computer, which is based on kinetic theory transport coefficients. The properties of hydrogen are present in the program in tabular form.

Heat fluxes, Q, are products of appropriate heat transfer coefficients, areas, and temperature differentials. Gas-side temperature differentials are the difference between combustion gas adiabatic wall temperatures and gas-side wall temperatures.

The adiabatic wall temperature was taken to be 90% of theoretical combustion temperature throughout. For mixture ratio 6:1 LOX/LH_2, the adiabatic wall temperature was taken as 4,954°F.

1. DeLaval Nozzle Regeneratively Cooled with Fuel

Design requirements for a regeneratively cooled deLaval type nozzle are shown in Figure 1. One hundred percent of the fuel is required for coolant. Gas-side or maximum expected temperatures in the tubing walls are shown in Figure 2. Figure 3 shows gas-side heat transfer coefficients which were used in arriving at the recommended design. The entrance pressure of the

coolant was taken as 1,300 psi, and the final pressure as given by the computer is 1,125 psi. The required injector face pressure was given as 1,050 psi. The design provides a margin of 75 psi for turn-around and other pressure losses.

Smaller tubes increase cooling effectiveness of the fuel, but they could not be used because excessive pressure losses resulted.

The hydrogen is heated from 40°R to 135°R in the tubes. From Figure 2, if the injector design does not allow for the excess temperature in the entrance section, some insulation in that area will be required.

The same general type of design, except with 50% of the fuel used as coolant and 4 mils of Rokide Z in the insulated region, could also be achieved; however, because the pressure drop for 100% of fuel coolant was within limits, the advantages of this system apparently outweighed resultant injector design complexity for allowance of different fuel densities.

2. DeLaval Nozzle Regeneratively Cooled with a Combination of Fuel and Turbine Gases

Turbine power is supplied by 16 gas generators furnishing 146 lb/sec of combustion exhaust gases of mixture ratio 1:1, LOX/LH_2. The theoretical combustion temperature of this mixture is 1,380°F, and the turbine exhaust pressure is approximately 100 psi. If these gases can be used to cool part of the nozzle exit section, the weight of the exit section can be reduced because thinner tubing can be used for construction.

Trials for total weight flow in both 3/4-in. and 1/4-in. tubes with 0.010-in. wall thickness were made in the General Heat Transfer Program of the digital computer, and the tube gas velocity was found to be excessive (that

is, supersonic) and the program failed to converge; however, the only compressible coolant that the program is presently capable of handling is pure hydrogen, and this was used to approximate the coolant flow. The use of hydrogen causes the velocity to be high by a factor of two.

Cursory hand work with the proper combustion gases indicated that wall temperature could be maintained below 1, 500°F in part of the exit section.

If interest is shown in this design, a more comprehensive digital program will be needed and more detailed analyses will be made as to whether cooling by this method with or without tube coating is possible.

3. Forced-Deflection Nozzle, Heated Hydrogen Cycle

It was determined that if 15% of the engine hydrogen can be heated to 1, 000°R and delivered at 1, 000 psi to the turbine, gas generators can be eliminated.

The eight-chamber forced-deflection engine was analysed to determine whether hydrogen at this temperature can be obtained. A possible design is shown in Figure 4. The design is such that the turbine supply is the total requirement for cooling the nozzles. For the design shown, the coolant attains a temperature of 976°R. Identification of heat-transfer stations on the nozzle are shown in Figure 5. Gas-side wall temperatures are shown in Figure 6 and the rate of coolant temperature increase is shown in Figure 7. For computation of gas-side heat transfer coefficients, it was necessary to assume a gas expansion line. It was also assumed that the expansion ratio for each chamber is 55. The gas-side heat transfer coefficients which were used are shown in Figure 8.

If the hydrogen temperature falls below the required amount, additional tubes can be placed in the gas stream at chamber boundaries. The baffles could then be used to give structural stability in the exit section.

Digital computer results showed that the final pressure is 1,280 psi. The input pressure was taken as 1,300 psi. There is ample pressure differential to account for turn-around and turbine line pressure losses. The delay in time in obtaining heated hydrogen is 0.35 sec.

III. CONVECTIVE FILM COOLING

In a literature survey, it was found that theory and experiments for film cooling are generally divided into two categories, depending upon whether the film coolant is a liquid or a gas[1]. Considerable work has been done at Purdue University with liquid films and this work has been summarized by A. Graham and M. Zucrow[2], wherein they compare the Purdue results with others in the field and make recommendations as to which expressions should be used for evaluating liquid coolant requirements.

A method for computing gaseous film coolant requirements has been presented by J. E. Hatch and S. S. Papell[3] of the National Aeronautics and Space Agency. They compared their data with information from experiments conducted by other organizations and the correlation was very favorable.

1. Fluke, G.L., Use of Convective Film Cooling in Large Booster Nozzles, Aerojet-General Corp., Report No. TTF-PN-61-5-1, May 1961.
2. Graham, A.R. and Zucrow, M.J., Film Cooling, Its Theory and Application, Purdue University Rocket Laboratory, No. TN-57-3, October 1957.
3. Hatch, J.E. and Papell, S.S., Use of a Theoretical Flow Model to Correlate Data for Film Cooling or Heating an Adiabatic Wall by Tangential Injection of Gases of Different Fluid Properties, Lewis Research Center, NASA, TN-D-130, November 1959.

Basically, the approaches in both liquid and gaseous film cooling are the same. Heat input (the product of a heat transfer coefficient and the temperature difference between the combustion gases and the film) is equated to the increase in heat content of the film coolant. For liquid coolants, the increase in heat content is absorbed in heating and vaporizing the coolant (at the liquid surface, $c_{pv}(\Delta t) = 0$) and for gaseous coolants, it is absorbed by increasing the temperature of the coolant gas. Evaluation of the heat transfer coefficients and appropriate temperature differences for heat input computations are the main concern in determining film coolant requirements.

It was previously determined that hydrogen is a poor liquid film coolant for large boosters because of the critical point of hydrogen (P_c = 188 psi T_c = 60°R), above which the heat of vaporization disappears, and its low density (sp gr = 0.07). Therefore, discussion of its use as a film coolant will be restricted to consideration as a gaseous coolant only. It is an attractive gaseous coolant from the standpoint of a heat sink because of its high heat capacity ($c_p \sim 3$ Btu/lb-°F).

A. As mentioned, gaseous film cooling uses the heat capacity of the gaseous coolant as a heat sink. In the expressions developed by J. Hatch and S. Papell[4] the heat input into the gaseous sublayer is:

$$h\,(T_{aw} - \bar{t}\,),$$

where h is the convective heat transfer coefficient, T_{aw} is adiabatic wall temperature of the combustion gases, and $\bar{t}$ is the average coolant temperature. The heat transfer coefficient, h, is given by

$$h = 0.0265\,(k_f/D_h)\,(Re_f)^{0.8}\,(Pr_f)^{0.3} \qquad \text{(Eq 3)}$$

4. Ibid

where Re_f is Reynolds number, k is combustion gas thermal conductivity, and D_h is hydraulic radius.

The subscript f refers to property values evaluated at the arithmetic mean of the static gas temperature, t_g, and the coolant slot exit static temperature.

B. In the derivation of the rate of heat absorption, a flat plate was used as the cooled model. The increment of cooled area was $L\Delta x$ and the assumptions are as follows:

1. The coolant film exists as a discrete layer (no mixing).
2. The temperature profile in the coolant film does not change rapidly with x.
3. The temperature gradient through the coolant is small.
4. No heat is conducted through or along the wall.
5. Conditions are uniform in the y direction (in any plane of flow).
6. The gas traverses the distance x' before heat diffuses through the gas layer to raise the wall temperature.

C. The heat balance for the incremental distance Δx along the plate may be written as

$$(\dot{w}\, c_p)_c\ (\bar{t}_2 - \bar{t}_1) = Ah\ (T_{aw} - \bar{t}) = L\Delta x\, h\ (T_{aw} - \bar{t}) \qquad \text{(Eq 4)}$$

By using the assumption B, 2 and 3, the following substitutions may be made in the above equation:

$$t_2 - t_1 \cong t_{w,2} - t_{w,1} \quad \text{and} \quad \bar{t} \cong t_y$$

the result is

$$\left(\frac{t_{w_2} - t_w}{\Delta x}\right)_{\lim \Delta x \to 0} = \frac{dt_w}{dx} = \left[\frac{hL}{(\dot{w} c_p)_c}\right] (T_{aw} - t_w) \qquad \text{(Eq 5)}$$

because $T_{aw} \neq f(x)$, this may be written as

$$\left(\frac{1}{T_{aw} - t_w}\right) d(T_{aw} - t_w) = -\frac{hL\, dx}{(\dot{w} c_p)_c} \qquad \text{(Eq 6)}$$

which, upon integration between limits gives

$$\ln\left[\frac{T_{aw} - t_w}{T_{aw} - t_c}\right] = -\left[\frac{hL(x - x')}{(\dot{w} c_p)_c}\right] \qquad \text{(Eq 7)}$$

or

$$\frac{T_{aw} - t_w}{T_{aw} - t_c} = \eta = \exp\left[-\frac{hL(x - x')}{(\dot{w} c_p)_c}\right] \qquad \text{(Eq 8)}$$

The quantity x' represents the distance that the coolant flows along the wall before the wall temperature increases. For values of $x \leq x'$ the wall temperature has not increased above the coolant temperature so that $\eta \leq 1$ for $x = x'$.

Factors affecting the length of x' are derived from one-dimensional heat conduction laws. For conduction with heat convection as a boundary condition, the following relationship holds:

$$\frac{\alpha_c}{S^2}\,\theta = \frac{C}{f\,(hS/k_c)} \qquad \text{(Eq 9)}$$

where α_c = thermal diffusivity ($k/\rho\, c$) of the coolant,

θ = time, S = coolant slot height or coolant layer thickness, and C = constant.

The value of x' may be determined for the flowing coolant gas.

$$x' = V_c\,\theta = C\left(\frac{S^2 V_c}{c}\right) f\left(\frac{1}{hS/k_c}\right) \qquad \text{(Eq 10)}$$

Data from the NASA film-cooling facility showed that the above equation can be expressed as:

$$x' = 0.04\left[\frac{S^2 V_c}{\alpha_c}\right]\left[\frac{1}{hS/k_c}\right] \qquad \text{(Eq 11)}$$

This, when substituted into Equation 8, yields

$$\ln \eta = -\left[\frac{hLx}{(\dot{w} c_p)_c} - 0.04\right] \qquad \text{(Eq 12)}$$

The thermal diffusivity coefficient for the coolant is evaluated at the coolant slot exit static temperature.

In obtaining correlations of data with the above equation, it was necessary for Hatch and Papell to include some additional factors. The final expression is

$$\ln\eta = -\left[\left(\frac{h\,L\,x}{(\dot{w}c_p)_c}\right) - 0.04\right]\left[\left(\frac{SV_g}{\alpha_c}\right)^{0.125} f\,(V_g/V_c)\right] \qquad \text{(Eq 13)}$$

which applies for

$$\frac{h\,L\,x}{(\dot{w}c_p)_c} \geq 0.04$$

V_g = gas velocity and V_c = coolant velocity.

The quantity $f\,(V_g/V_c) = 1 + 0.4\,\tan^{-1}\left(\frac{V_g}{V_c} - 1\right)$ when $V_g/V_c \geq 1.0$ and

$f\,(V_g/V_c) = \left(\frac{V_c}{V_g}\right)^{1.5(V_c/V_g - 1)}$ when $V_g/V_c \leq 1.0$

The angle in the above equation is in radians.

The gas velocity V_g is used in place of V_c in SV_c/α (velocity ratio of 1:1) and $f\,(V_g/V_c)$ corrects for velocity ratios that differ from 1:1.

$$\eta = 1 \text{ for } \left(\frac{h\,L\,x}{(\dot{w}c_p)_c}\right) \leq 0.04$$

If equation 14 is solved for the coolant flow rate, the result is

$$\dot{w}_c = \frac{h\ L\ x}{c_{p_c}} \left[\frac{1}{\left(\frac{-\ln \eta}{\left[SV_g/\alpha_c \right]^{0.125} f\ (V_g/V_c)} \right) + 0.04} \right]$$

(Eq 14)

In addition to Equation 14, the continuity equation for coolant flow also must be satisfied. It is

$$S = \dot{w} \rho_c V_c L \qquad \text{(Eq 15)}$$

Correlations were obtained by Hatch and Papell with both air and helium as the film coolants. Because they have widely varying physical properties, it was recommended that the expressions be used for any gaseous coolant. Tangential injection is the only restriction.

As the ratio of V_g/V_c departs from unity, the film coolant requirements increase. Figure 9 is a representation of the amount of film coolant required as a function of V_g/V_c. It is seen that the film requirements increase rapidly when the film velocity exceeds the gas velocity. The requirements also increase when the gas velocity exceeds the film velocity, but at a slower rate. The film coolant requirements remain constant when the coolant velocity is 2% (or less) of the gas velocity.

Although all of the theoretical developments for both liquid and gaseous film were based on the assumption that no heat is absorbed by the duct wall when covered by a film, Kinney and Sloop[5] in their experiments found

5. Graham, A.R. and Zucrow, M.J., op. cit.

that the temperature of the outside duct wall rose to the boiling point of the liquid when covered by a liquid film. These results indicate that the present film theory correlations are a special case of an extended boundary value problem of transient heat conduction in a duct wall, and that the duct wall heat capacity can be used as an additional heat sink.

The foregoing expressions were used for evaluation of hydrogen film coolant requirements for the two-million-pound Laval chamber. Because of the similarity in Equations 1 and 3, values of gas-side heat transfer coefficients which had been evaluated for regenerative cooling requirements by Equation 1 and which are presented in Figure 2 were used for evaluation of film coolant requirements.

The maximum allowable wall temperature was set at 1,500°F, and the wall was assumed to be made of sheet stock metal. The wall thickness requirements were determined by taking the allowable stress of 347 SS and 310 SS at 1,500°F as 1,500 psi. The weight of such a nozzle is approximately 3,200 lb. This could be decreased by using pressurized steel tubing, (the weight of tubing in the Laval design of Figure 1 is approximately 1,700 lb). Utilization of a nickel base alloy, Rene 41, for nozzle construction would cut the weight to 534 lb. Recent tests performed in the Materials Research Laboratory at Aerojet-General Corporation[6] determined that its stress limit at 2% yield is approximately 95,000 psi at 1,400°F. It costs approximately three times as much per pound as stainless steel.

Because the wall properties are not included in the calculation of film requirements, the wall material is independent from this standpoint.

6. Flins, F.J., et al, Materials and Fabrication Development Program for the -5 Titan Engine, Aerojet-General Corporation Report No. RM-5940, February 1961.

Previous work at Aerojet-General Corporation has shown that injectors can be designed so that they provide film cooling for the combustion chamber for approximately 60% of the converging nozzle section. Based on this, cooling requirements were computed for the two-million-pound engine with coolant injectors assumed to be located at expansion rates 1.55 upstream of the throat and expansion ratio 2.89 downstream of the throat. For this calculation, it was assumed that the coolant thickness adjusts itself so that the velocity of the coolant remains at combustion gas velocity. If the coolant velocity exceeds the gas velocity, its effectiveness falls off sharply (see Figure 9).

Results of calculation showed that 105 lb/sec and 102 lb/sec of coolant are required at the upstream and downstream injection points, respectively. The total film coolant requirement of 207 lb/sec represents 25% of the total fuel flow rate. Aside from the computed requirements, the practicability of film cooling with hydrogen is unknown at the present time.

Scientists at the Applied Physics Laboratory[7] in 1959 reviewed the literature for film cooling and suggested areas of investigation for future research. They pointed out in detail many of the uncertainties connected with it.

A successful film-cooled engine was developed by Aerojet-General Corporation in 1959[8]; however, the film coolant was liquid.

7. A Survey of Basic Scientific Problems Associated with the Behavior of Materials at High Temperatures, Areas VI and VII, John Hopkins University Applied Physics Laboratory, TF-333-1, June 1959.
8. Sachs, S. L., Liquid Propellant Rocket Engine for Sled Propulsion Use, Aerojet-General Corporation Report No. 0214-01Q-2, June 1959.

IV. UNCOOLED NOZZLES

A. Uncooled or so-called radiation-cooled nozzles constructed of a radiating material are often proposed as a solution to nozzle cooling problems.

Basically, this approach is that the heat flux is to be transferred from the nozzle by means of radiation. That is,

$$(h_g + h_r)\ (T_g - T_{wg})\ A_i = \epsilon \sigma\ T_{wc}{}^4 A_o$$

For a thin shell of large radius it can be assumed that A_i and A_o (inside and outside surface areas, respectively) and T_{wg} and T_{wc} are equal. The expression, then, is

$$(h_g + h_r)\ T_w = \epsilon \sigma\ T_w{}^4.$$

The maximum heat that can be radiated occurs when emmissivity is unity.

The equilibrium wall temperature can then be evaluated as a function of heat transfer coefficient. This has been done for the two-million-pound deLaval nozzle and equilibrium radiation wall temperatures as functions of expansion ratio are shown in Figure 10.

It is seen that all temperatures exceed the allowable for materials which are being considered for the two-million-pound LOX/LH_2 engine system.

The use of composite uncooled combustion chamber and nozzle walls is a field of constant endeavor in solid rocketry. Basically, the construction depends upon steel for the strength material which is protected thermally by insulation. The insulation, in turn, is protected from erosion by being shielded from the gas by a high-temperature material, such as tungsten. Thickness requirements of each material are based upon a transient heat-transfer analysis which meets the requirements of the given design. The usual duration of solid rockets is 60 sec

or less. The requirements for the large boosters are for 180 sec. Design of such nozzles for large boosters would be comprised of thickening wall materials so that thermal limitations are not exceeded. Their successful operation would then be dependent upon keeping such a design intact for full duration firing of the booster. Liquid engines have large sources of coolant aboard, the use of which represents a tremendous advantage in nozzle reliability.

Ablative cooling as the name implies refers to the planned removal of material from a particular region. The removal is dependent upon its absorption of heat.

The analytical treatment of the process of ablation is made by a heat balance. Some manufacturers of ablative materials rate their products in terms of equivalent heats of ablation. The heat flux into the ablative material is then computed and the required amount estimated.

Other more mathematically complex treatises have been made in an attempt to predict the behavior and thickness requirements of ablative materials.

A recent report[9] by the Aerojet-General Corporation was concerned with ablative thrust chamber feasibility firings and devoted a section to the development of ablative theory, which was used in that study.

The ablative materials used were Refrasil and Pyrotex. The resin was assumed to evaporate, and the Refrasil or Pyrotex was assumed to char. Refrasil is a trade name for high-silica-glass fringe tape, and Pyrotex is a trade name for asbestos tape. Both materials are impregnated with phenolic silicone resin.

---

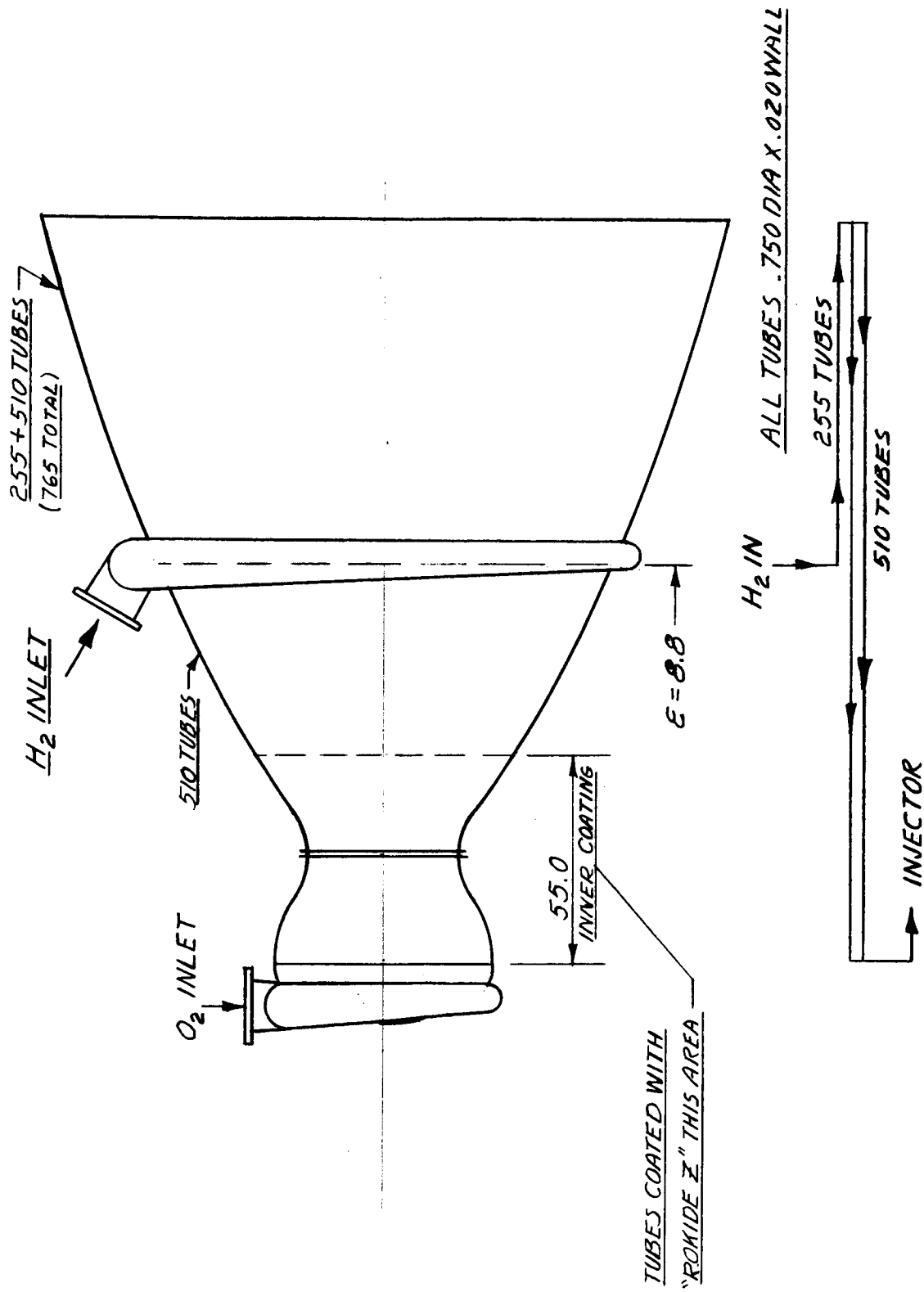
9. Research and Development on Components for Pressure-Fed Liquid Oxygen-Hydrogen Upper Stage Propulsion Systems, Task I, Contract AF 04(611)-5170 Report No. 1933 (Final). February 1961. Aerojet-General Corporation, Azusa, California.

The correlation of the results of firings with predicted thickness requirements were reported to be good. The work was based upon 65 psia chamber pressures.

Previous experience with uncooled resin ablative type chambers in solid rocketry of 1,000 psia chamber pressures have shown severe deterioriation, especially in the throat area of the chamber when the material was exposed to hot combustion gases. The erosion of material when the insulation leaves the chamber is practically impossible to predict.

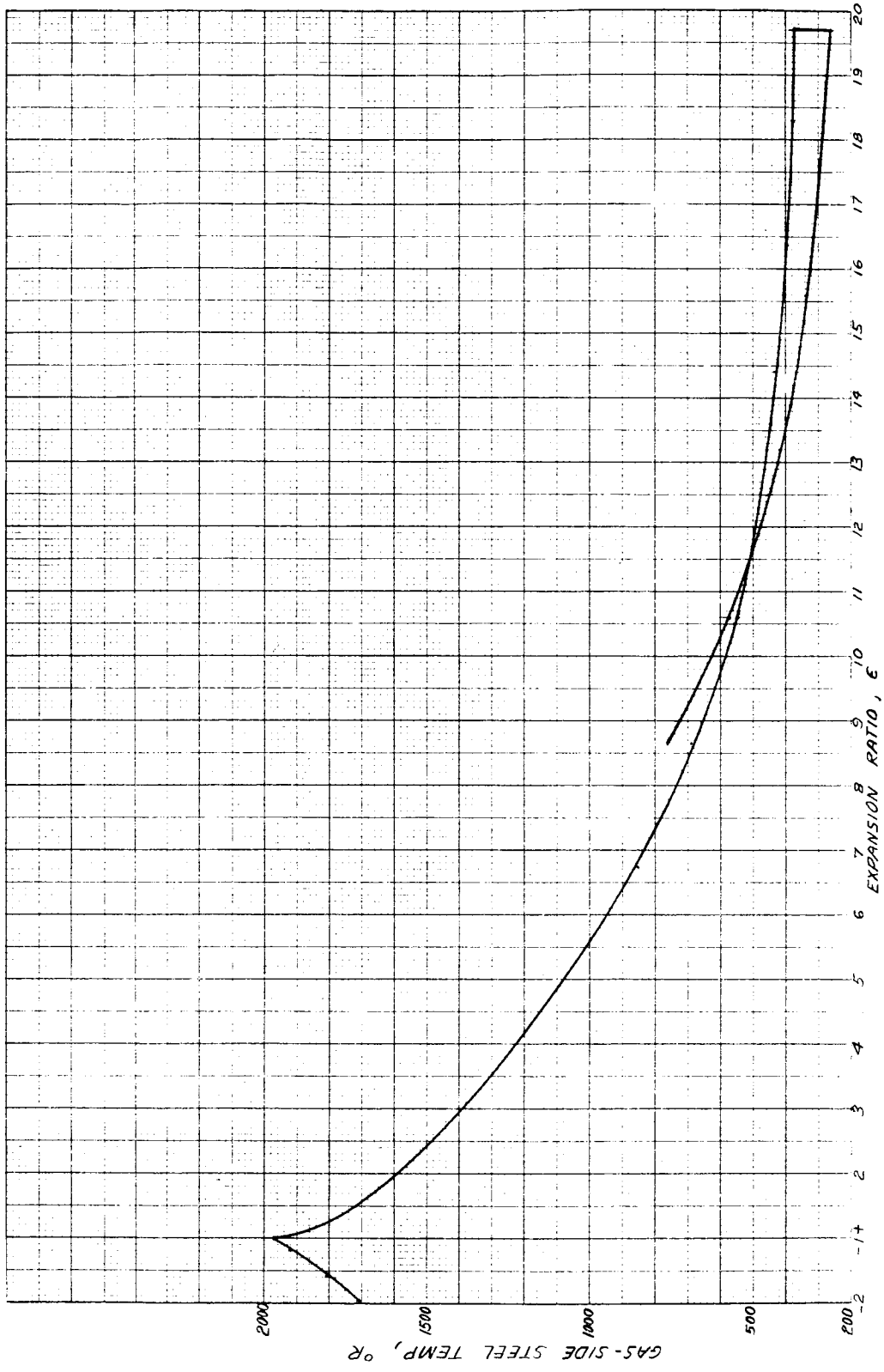
Ablative materials are very expensive presently. Their cost, coupled with the high pressures they must withstand, practically eliminates the consideration of resin-type ablative materials for large boosters. The Aerojet-General Corporation has a research program for types and possible uses of ablative materials, and the possibility of their use is, however, subject to continuing review.

(End text)



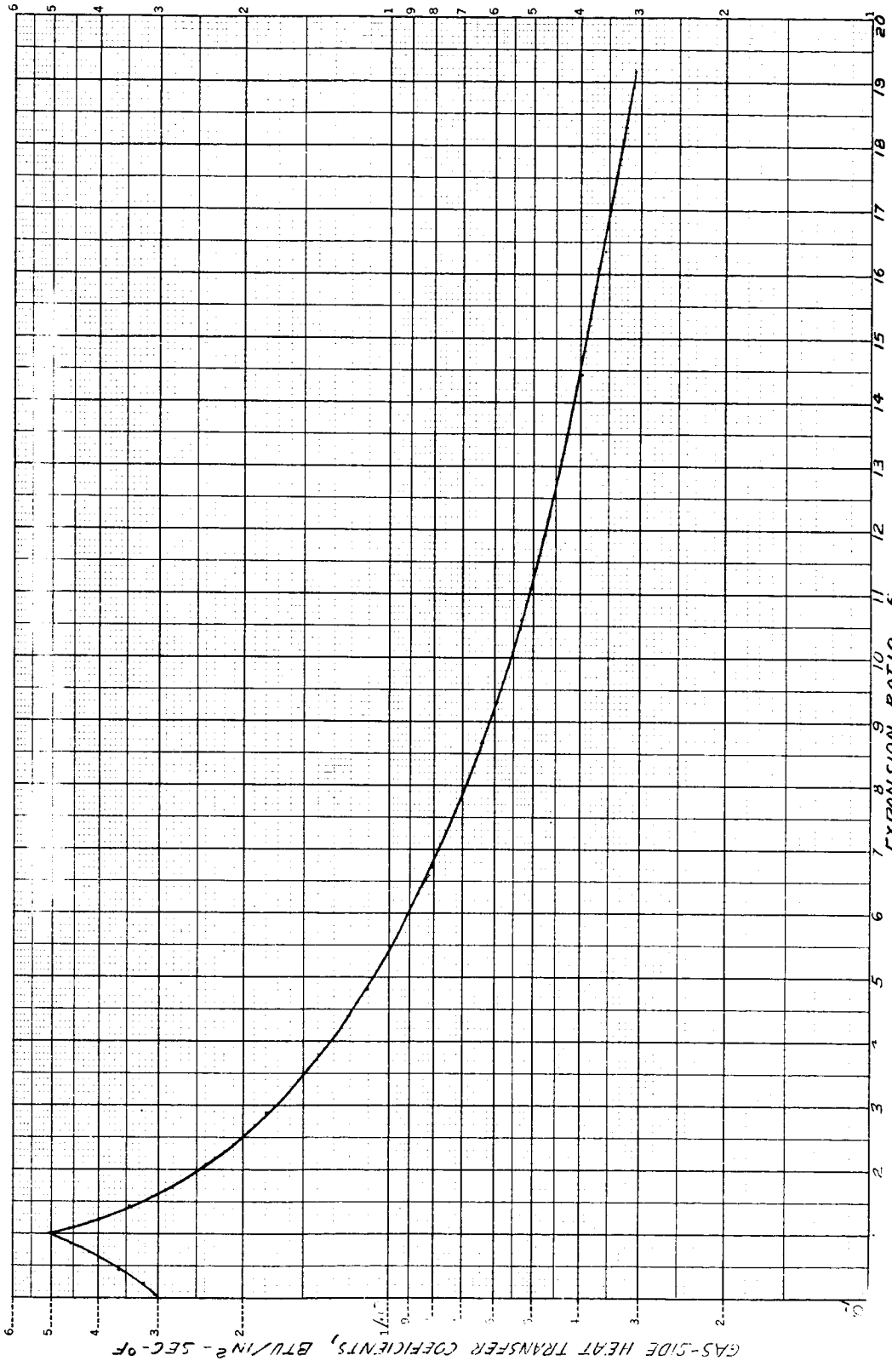
Regeneratively Cooled 2000K DeLaval Nozzle

Figure 1



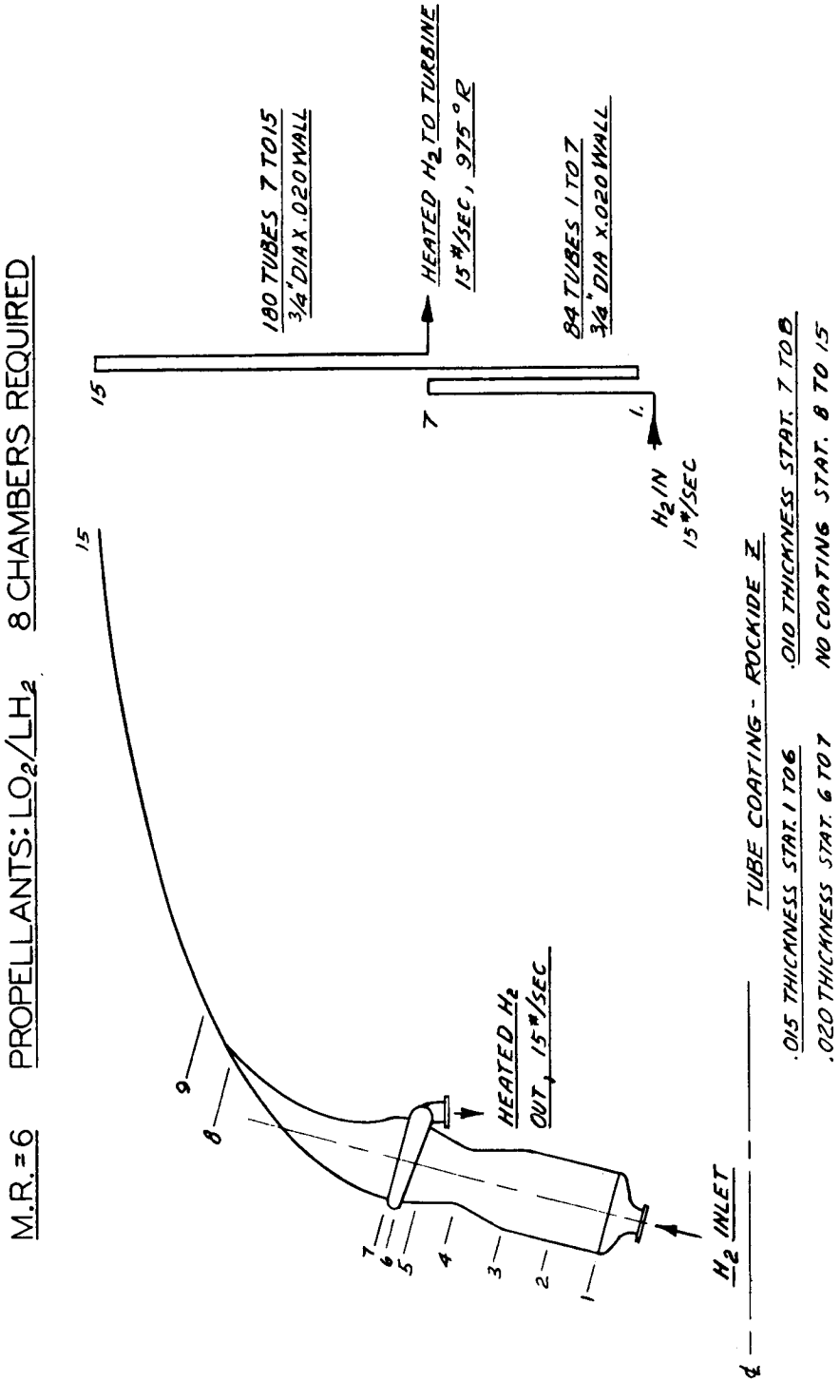
Regeneratively Cooled 2000K DeLaval Nozzle, Gas-Side Steel Temperatures

Figure 2



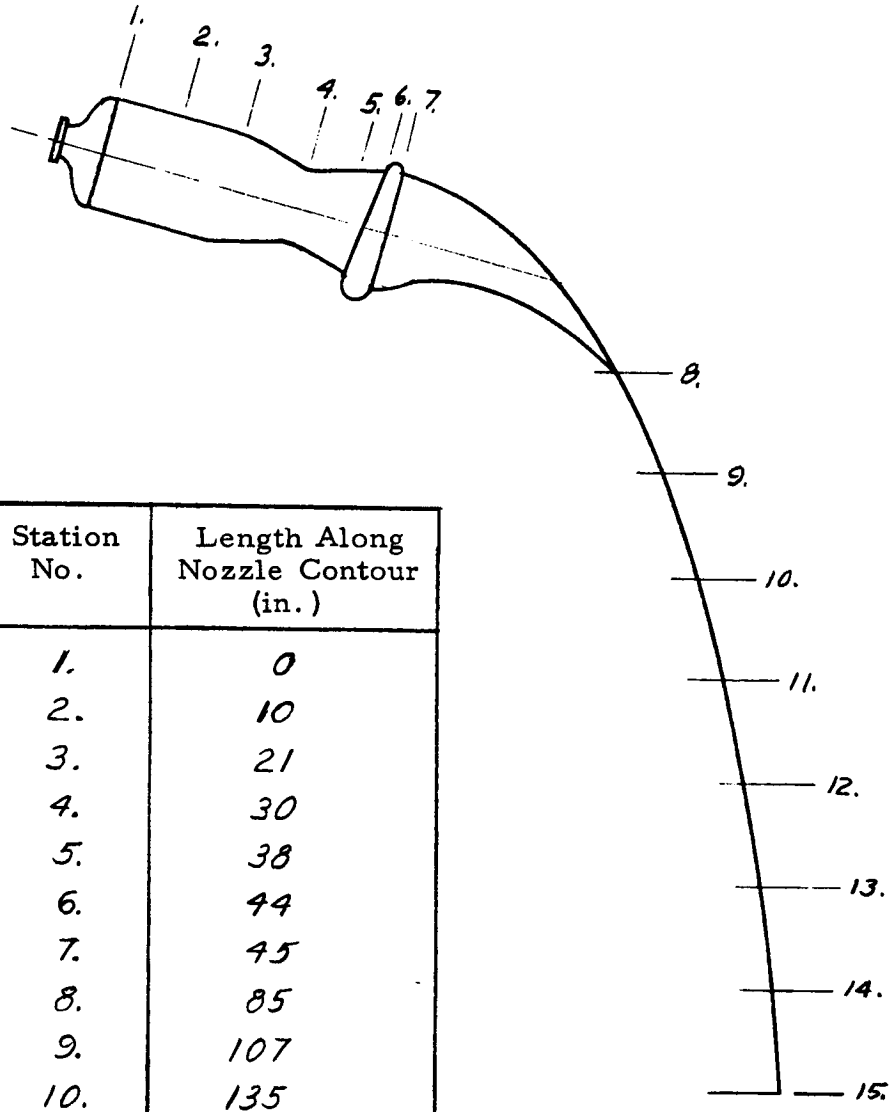
Regeneratively Cooled 2000K DeLaval Nozzle, Gas-Side Heat-Transfer Coefficients

Figure 3



Heat Exchanger Design Requirements for Heated Hydrogen Cycle-2000K Forced-Deflection Engine

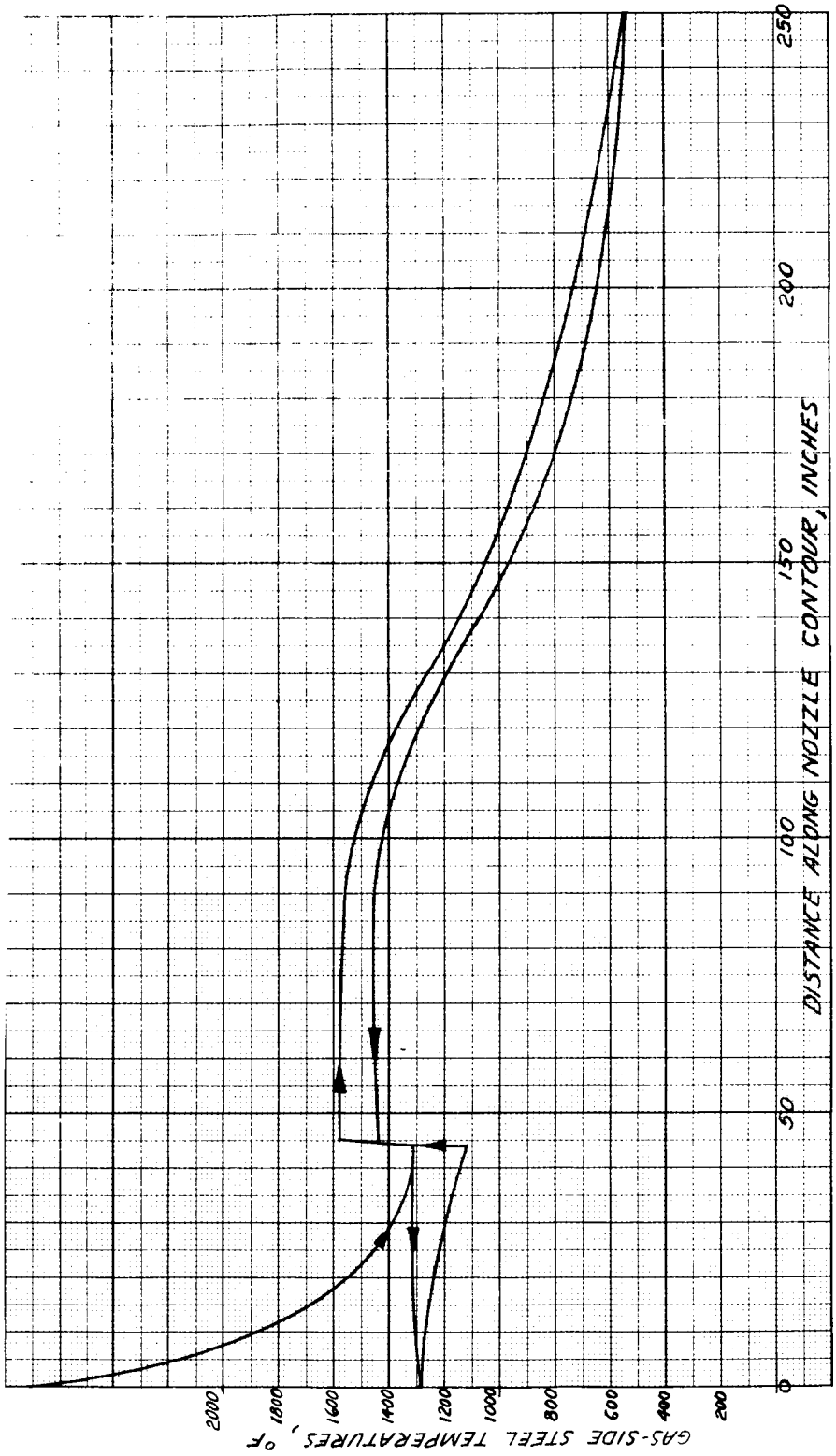
Figure 4



Station No.	Length Along Nozzle Contour (in.)
1.	0
2.	10
3.	21
4.	30
5.	38
6.	44
7.	45
8.	85
9.	107
10.	135
11.	157
12.	179
13.	204
14.	228
15.	252

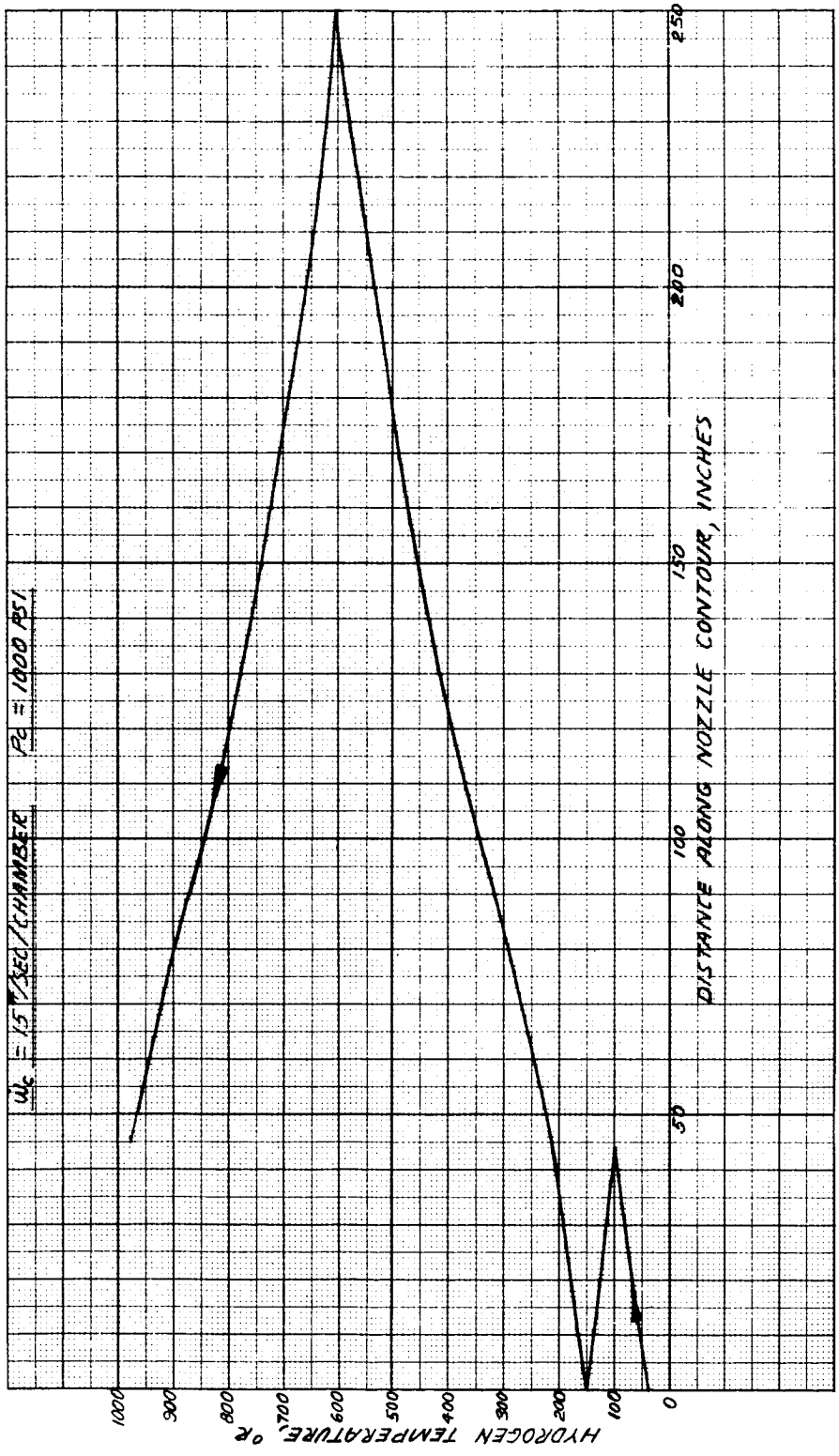
Heated Hydrogen Cycle 2000K Forced-Deflection Nozzle
Identification of Heat Exchanger Stations

Figure 5



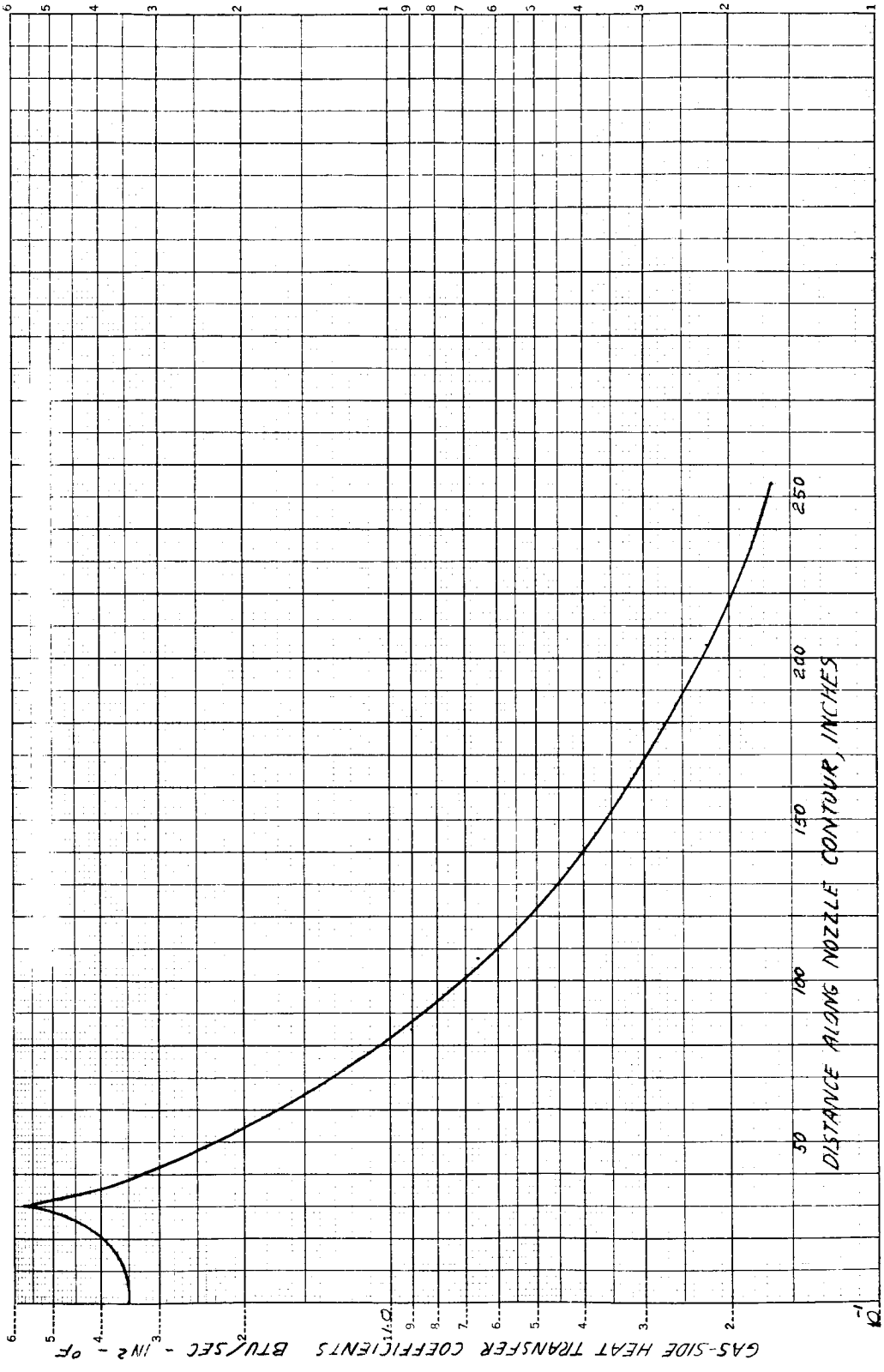
Heated Hydrogen Cycle 2000K Forced-Deflection Engine
Gas-Side Steel Temperatures

Figure 6



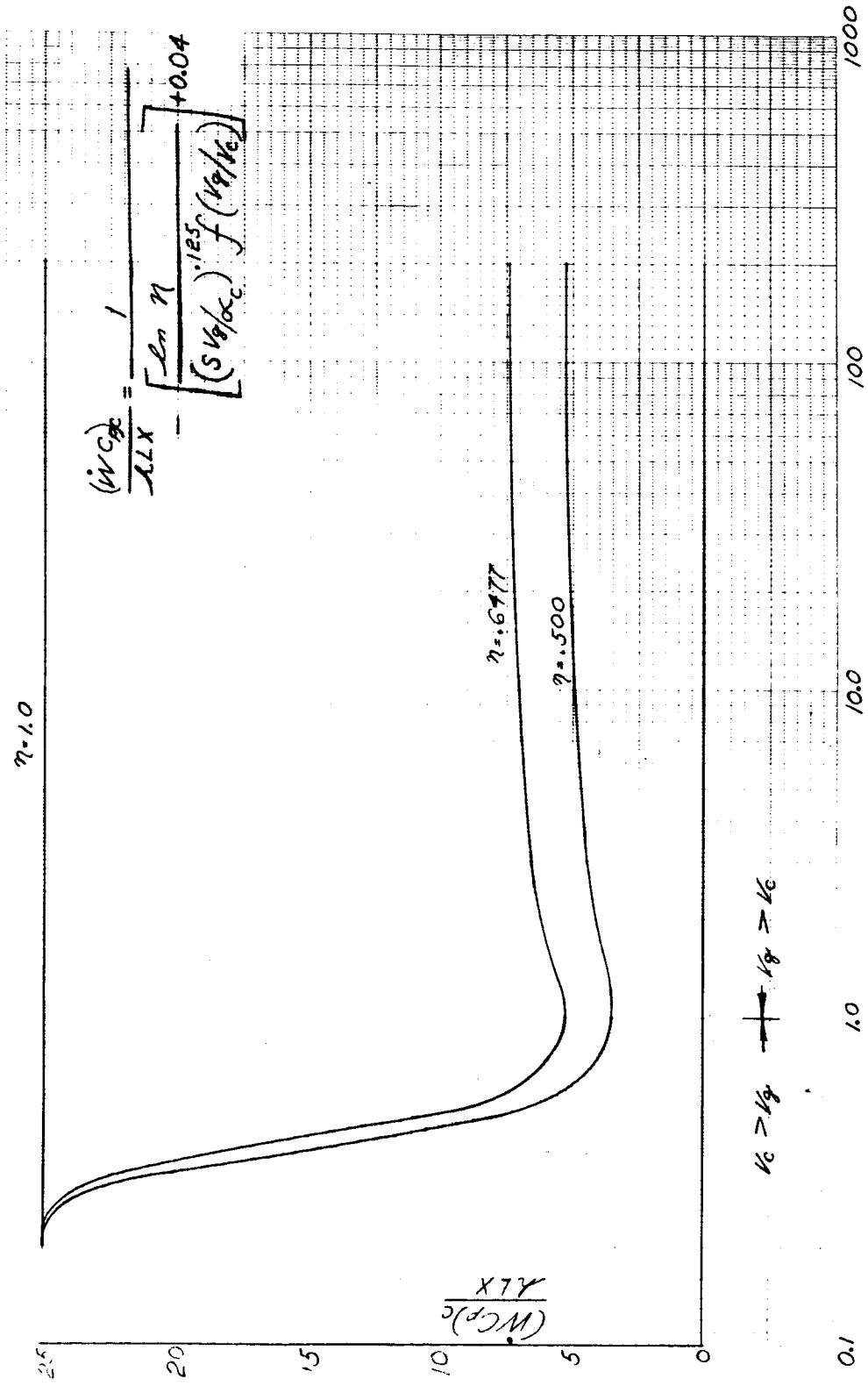
Heated Hydrogen Cycle 2000K Forced-Deflection Engine
Heated Hydrogen Temperature

Figure 7



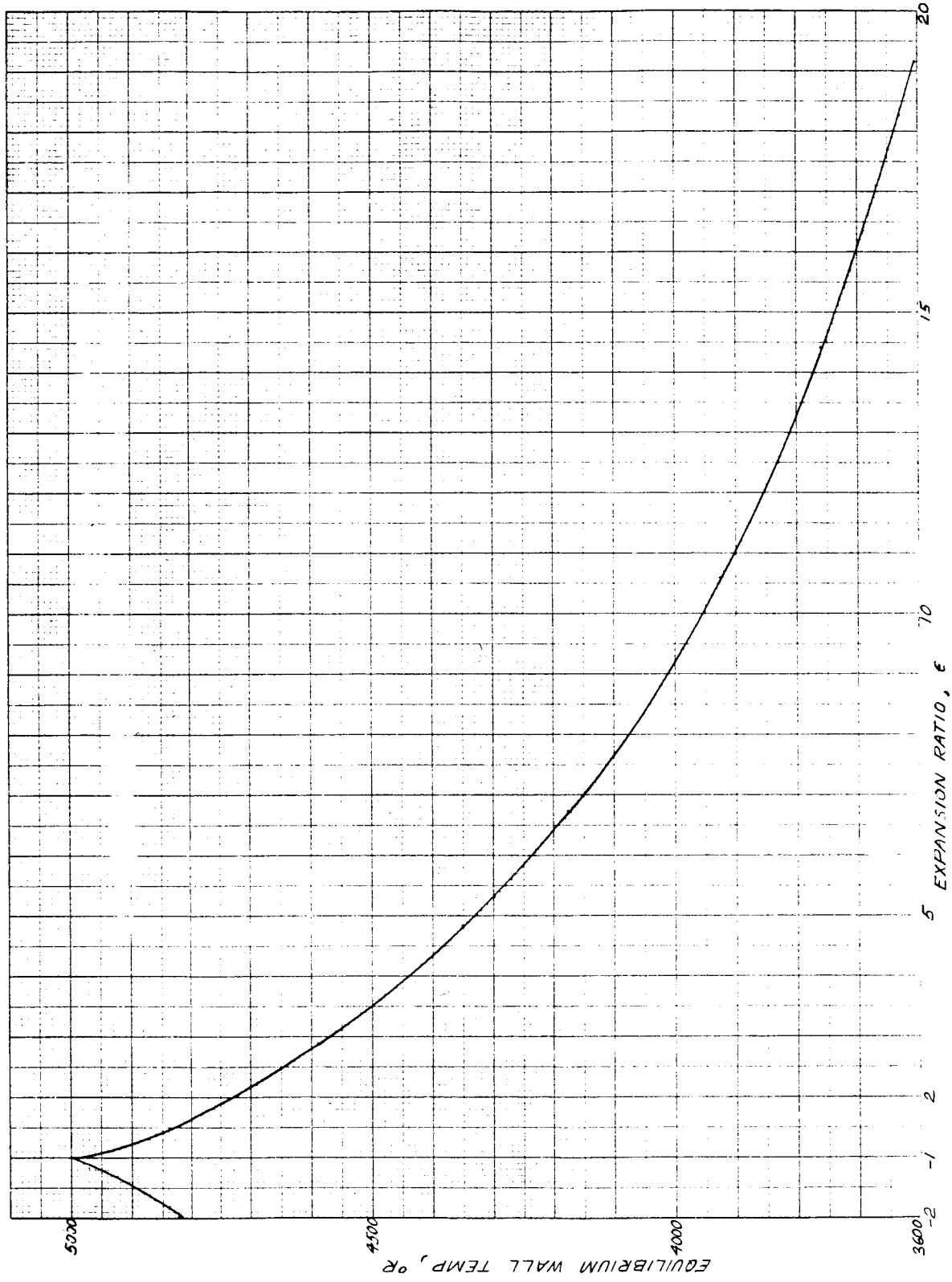
Heated Hydrogen Cycle 2000K Forced-Deflection Engine
Gas-Side Heat Transfer Coefficients

Figure 8



Required Gaseous Film Coolant vs Ratio of Gas to Coolant Velocities

Figure 9



2000K DeLaval Nozzle Radiation Cooled Equilibrium Wall Temperatures

Figure 10

APPENDIX B

COMPARISON OF COST CRITERIA

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	1
II.	CONVENTIONAL VEHICLES AND PROPELLANT COMPARISON	2
III.	RECOVERY METHODS	3
IV.	CONVENTIONAL VERSUS UNCONVENTIONAL SYSTEMS	4

FIGURE LIST

	FIGURE NO.
Conventional Vehicle and Propellant Comparison, Fixed Payload	1
Conventional Vehicle and Propellant Comparison, Fixed Thrust	2
Recovery Method Comparison, Fixed Payload	3
Recovery Method Comparison, Fixed Thrust	4
Comparison of Conventional vs Unconventional Configurations, Fixed Payload	5
Comparison of Conventional vs Unconventional Configurations, Fixed Thrust	6

I. INTRODUCTION

A cost comparison was established in the Task 1 effort, for various types of vehicles upon the basis of equal orbital and escape payloads.

These comparisons were instrumental in propellant selection, comparisons of recovery methods, and selection of the unconventional system. A similar comparison of vehicles has been completed upon the basis of equal sea-level thrust. It is the purpose of this discussion to present comparison results and show the significant differences in the results obtained from the two methods of vehicle cost comparison.

The cost curves, Figures 1, 3, and 5, for the fixed payload comparison are similar to those shown in Volume 2. The cost model and vehicle assumptions used in establishing these figures are the same, except for the following:

The conventional oxygen/hydrogen, first-stage vacuum specific impulse was changed from 411 to 420 lb-sec/lb.

The oxygen/RP-1, first-stage vacuum specific impulse was changed from 309 to 318 lb-sec/lb.

These increases in performance result because of a greater nozzle area ratio which is desirable in high velocity increment booster stage vehicles.

The vacuum specific impulse of the unconventional stage was changed from 456 to 451 lb-sec/lb.

An engine vacuum specific impulse of 456 was based upon a thrust chamber pressure of 3,000 psia and an area ratio of 190:1. Results of a parameter study during Task 2 indicate that a thrust chamber pressure of 2,500 psia with a

I, Introduction (cont.)

corresponding area ratio of 140:1 is probably closer to optimum; therefore, the engine specific impulse was changed to correspond to these new values of thrust chamber pressure and area ratio.

The payloads used in the fixed payload analysis are, 122,000 lb for the 300-nm-orbit mission and 48,900 lb for the escape mission. These payloads represent the capability of a 2-million-lb-thrust (sea level) two-stage conventional oxygen/hydrogen vehicle. All vehicles were sized and costed with these payload capabilities.

Second-stage development costs were assumed to be zero.

The fixed thrust cost comparison was made for vehicles with a sea-level thrust of 2-million lb and a takeoff weight of 1.6-million lb. These costs are presented in Figures 2, 4, and 6. The second-stage development costs are not included in the analysis.

II. CONVENTIONAL VEHICLES AND PROPELLANT COMPARISON

A comparison of conventional vehicles upon a fixed payload basis is delineated in Volume 2. The conventional vehicles considered are:

A. A two-stage oxygen/hydrogen vehicle.

B. A single-stage oxygen/hydrogen vehicle

C. A two-stage oxygen/RP-1 vehicle

D. An oxygen/RP-1 booster with an oxygen/hydrogen second-stage vehicle.

II, Conventional Vehicles and Propellant Comparison (cont.)

It is concluded in the referenced study that there is a little cost difference between the three vehicles except at 400 flights (where the single-stage-to-orbit oxygen/hydrogen vehicle begins to show a significant advantage over the other vehicles). This would favor the two-stage oxygen/hydrogen vehicle, with a booster of single-stage-to-orbit capability. The vehicle is also better than the other vehicles upon the basis of reliability and growth potential. Therefore, the use of oxygen/hydrogen propellant for future boosters is recommended.

The fixed thrust comparison, Figure 2, indicates the two-stage oxygen/hydrogen vehicle is significantly superior upon a cost basis to vehicles with oxygen/RP-1 stages. It can also be concluded that a conventional single-stage-oxygen/hydrogen to-orbit vehicle is inferior costwise to all other orbital vehicles using either oxygen/RP-1 or oxygen/hydrogen boosters; however, the single-stage-to-orbit has an advantage based upon reliability and growth potential. In addition, the single-stage capability can be designed into the oxygen/hydrogen booster with very little extra effort; therefore, it would still be desirable to have this capability for missions with smaller payloads.

III. RECOVERY METHODS

The following recovery methods were investigated.

A. Winged recovery

B. Recovery from a low booster velocity increment (11,000 ft/sec)

C. Single-stage orbital recovery.

III, Recovery Methods (cont.)

Conclusions presented in Volume 2 for the fixed payload comparison show there is little cost difference between the three recovery methods; however, upon the basis of reliability and growth potential, the recovery-from-orbit method is best; therefore, this method is considered the most promising recovery method for future boosters. Figure 3 shows the comparison for a fixed payload basis.

Based upon the fixed thrust cost curves (Figure 4), only two methods of booster recovery are competitive with the two-stage conventional nonrecoverable vehicle. These are the winged recovery at the higher launch rates, and recovery from a low ideal velocity increment (11,000 ft/sec) for an oxygen/hydrogen booster and oxygen/hydrogen second-stage vehicle. The latter vehicle was not shown in the fixed payload based costs because it was inferior to a similar vehicle except when using an oxygen/RP-1 booster for recovery.

The reliability of these two recovery methods would not be significantly different; however, the winged recovery vehicle would have the advantage of a greater growth potential.

IV. CONVENTIONAL VERSUS UNCONVENTIONAL SYSTEMS

The comparison of conventional and unconventional vehicles upon the basis of fixed payload is discussed in Volume 2. Figure 5 presents the costs upon a fixed payload basis and Figure 6 presents the costs on a fixed thrust basis. The major differences between the two comparison bases are:

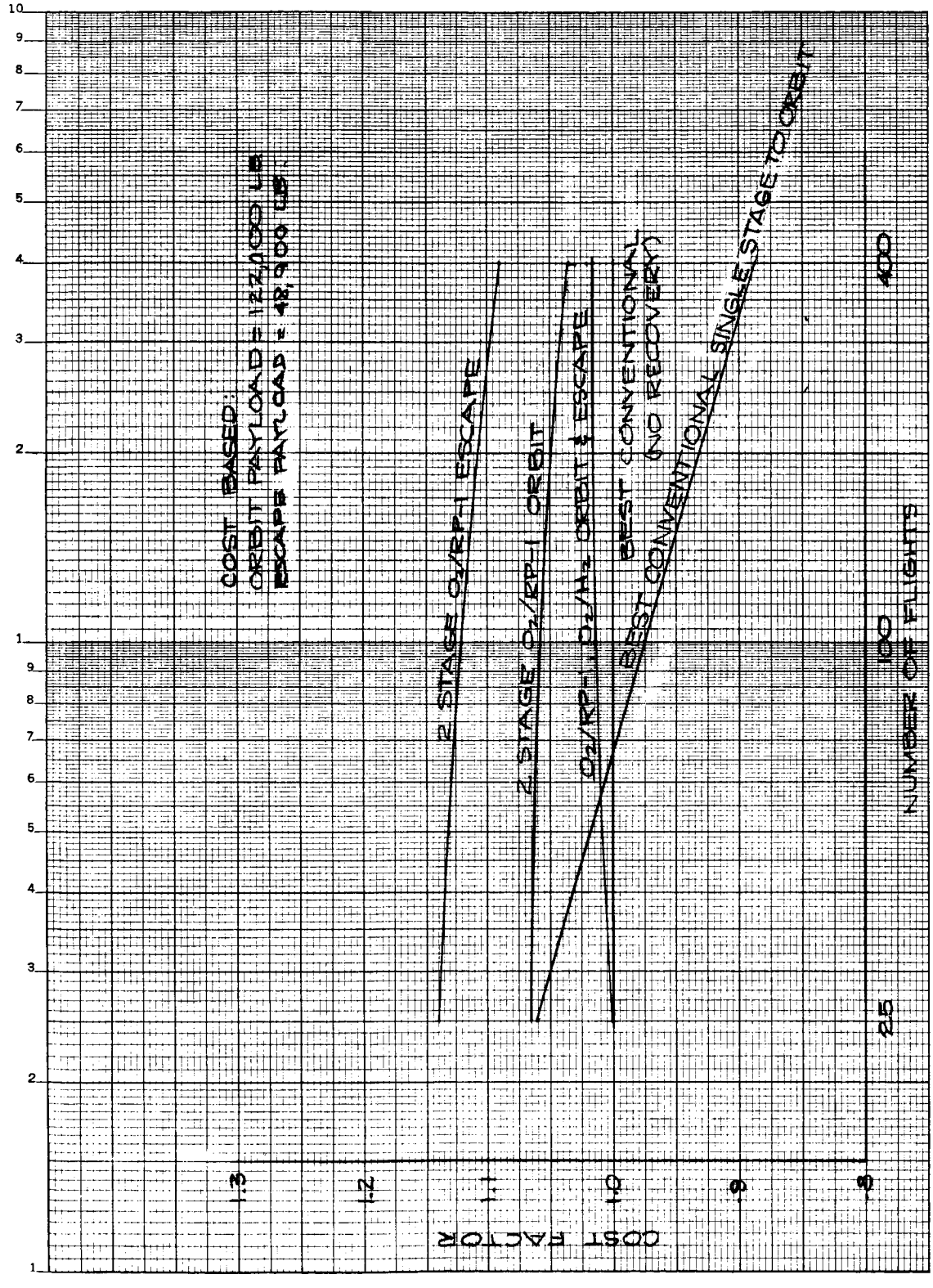
A. The recovery-from-orbit vehicles with unconventional boosters do not show an advantage over the two-stage conventional vehicles at equal thrust.

IV, Conventional Versus Unconventional Systems (cont.)

B. The best unconventional single-stage-to-orbit vehicle is inferior to the two-stage unconventional vehicle when compared at the same thrust; whereas it was superior to the two-stage unconventional vehicle on the fixed payload basis.

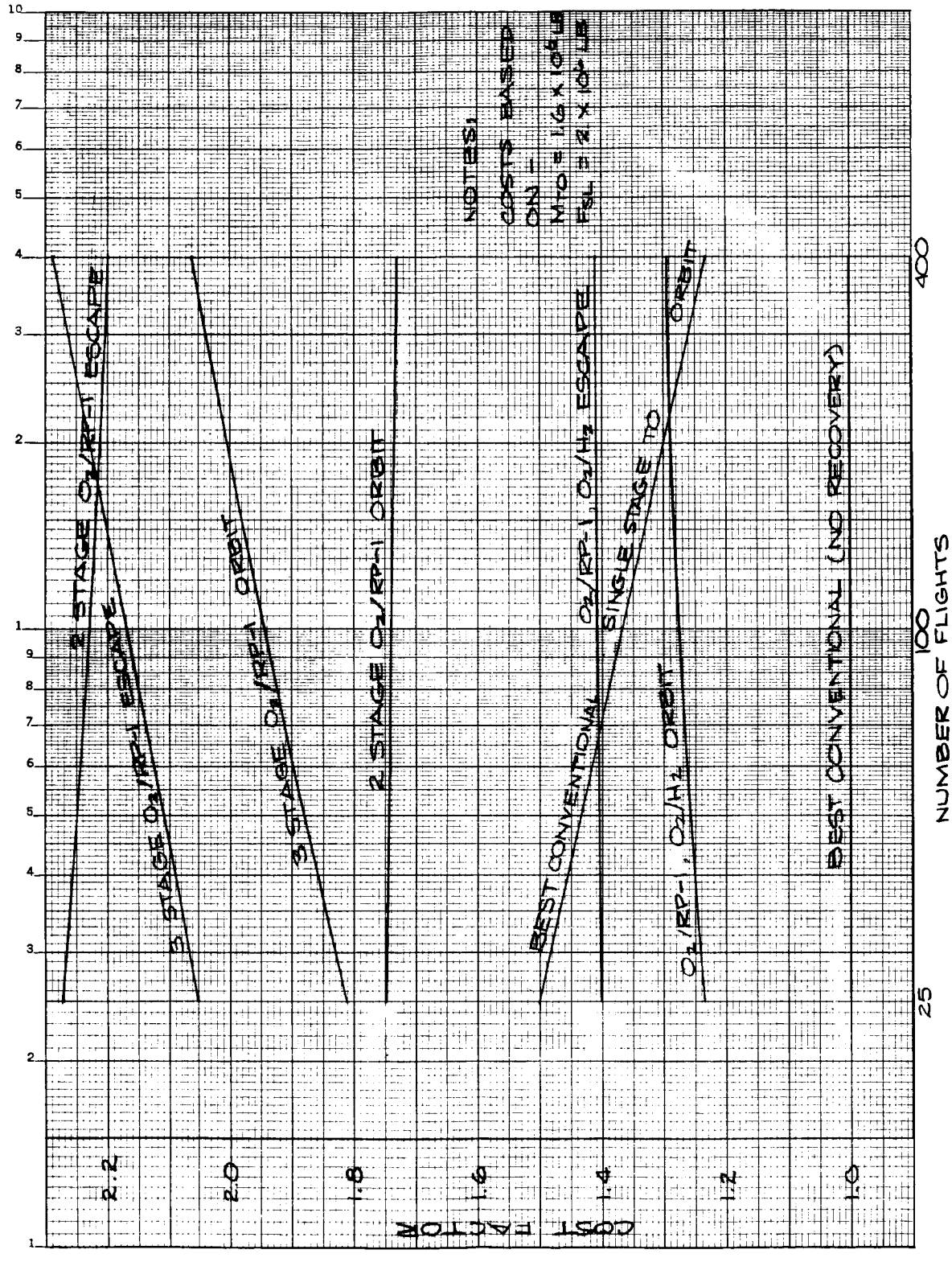
However, as noted, if the second-stage development cost is included in the cost comparison, the best unconventional single stage vehicle would be superior to the best two-stage unconventional vehicle. In addition, the single-stage vehicle would be more reliable and have greater growth potential.

It is concluded that the selection of a single-stage unconventional vehicle to orbit in Task 1 would not be altered by the different cost comparison method.



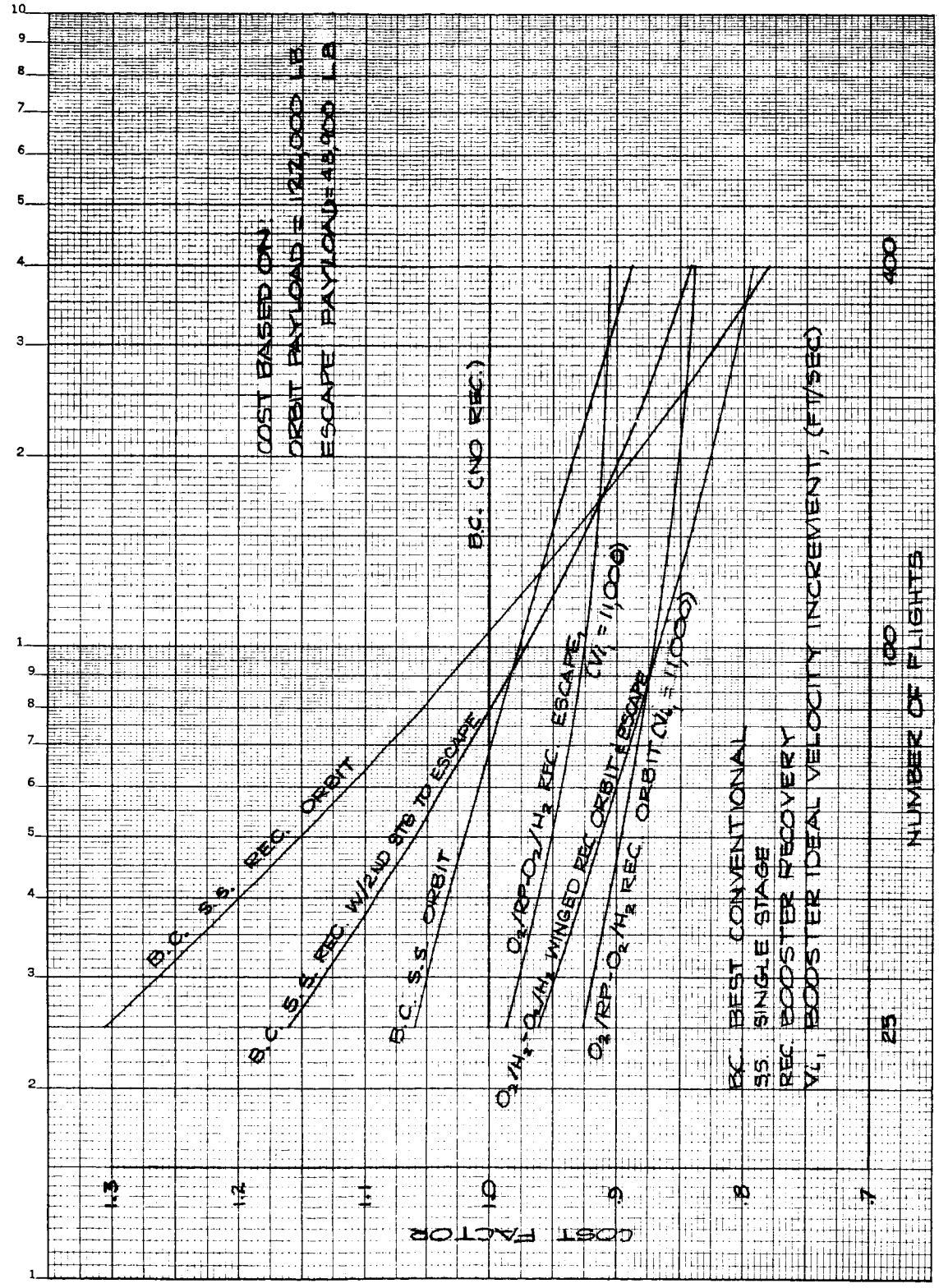
Conventional Vehicle and Propellant Comparison, Fixed Payload

Figure 1



Conventional Vehicle and Propellant Comparison, Fixed Thrust

Figure 2



Recovery Method Comparison, Fixed Payload

Figure 3

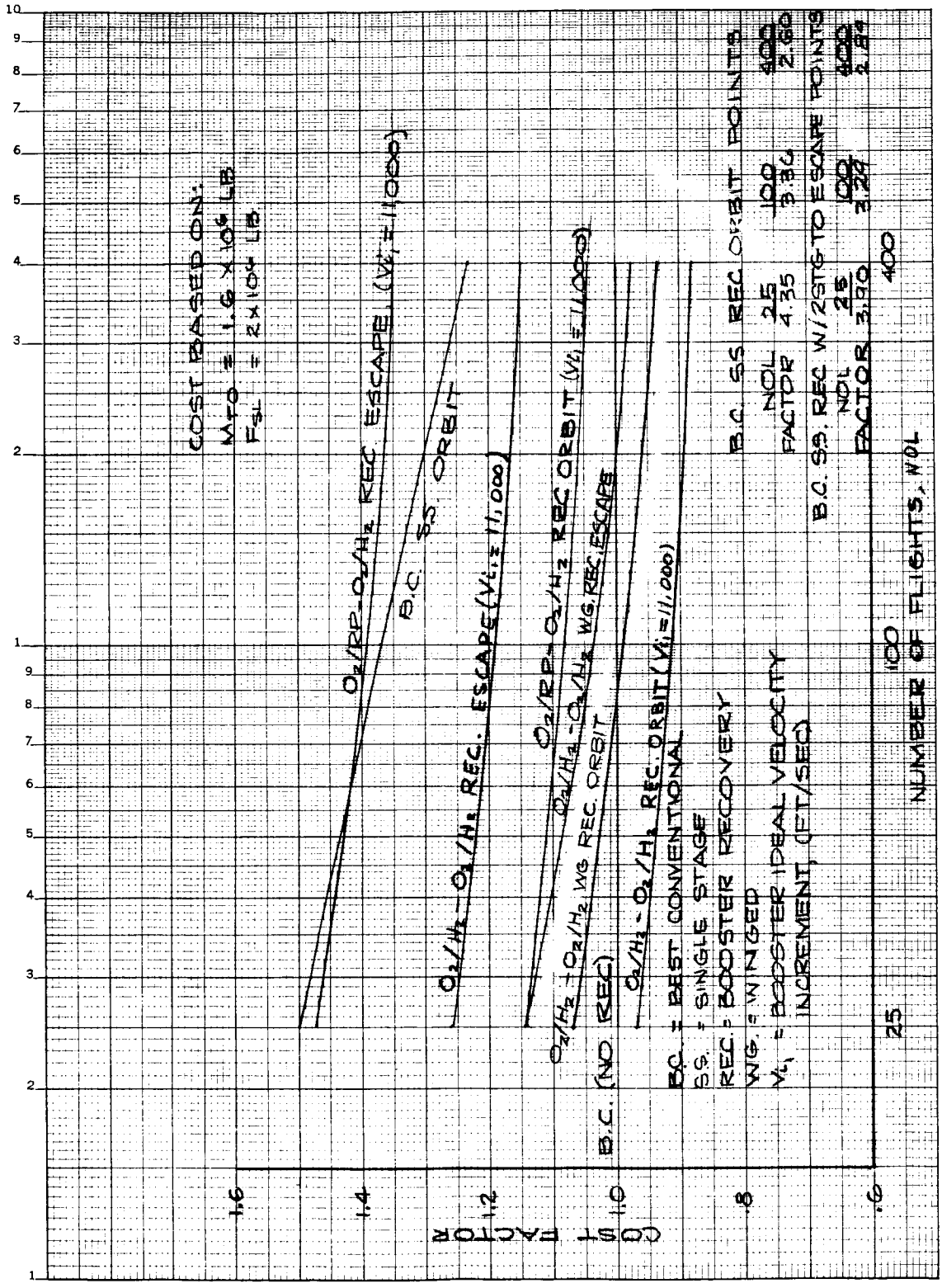
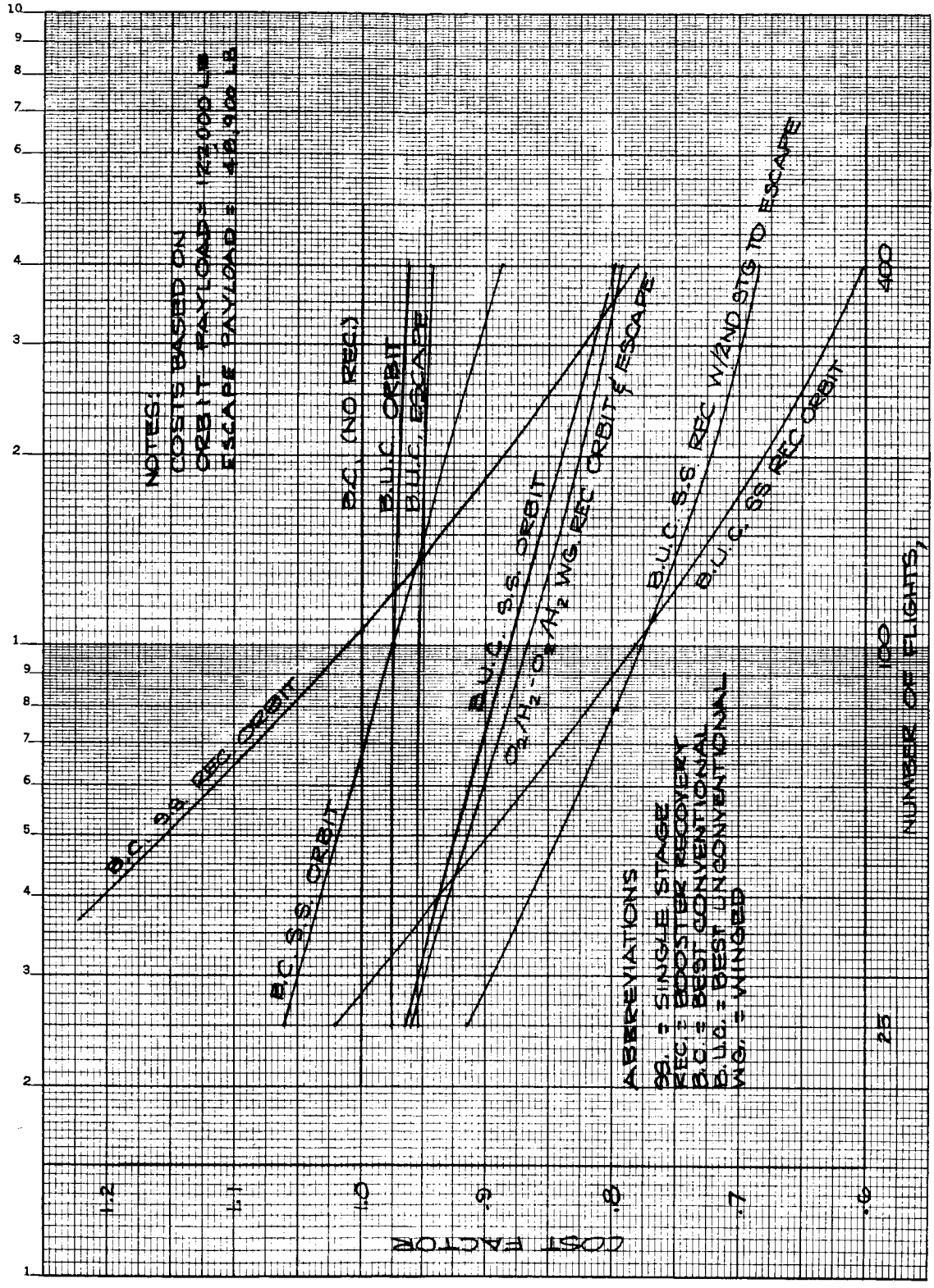


Figure 4

Recovery Method Comparison, Fixed Thrust



Comparison of Conventional vs Unconventional Configurations, Fixed Payload

Figure 5

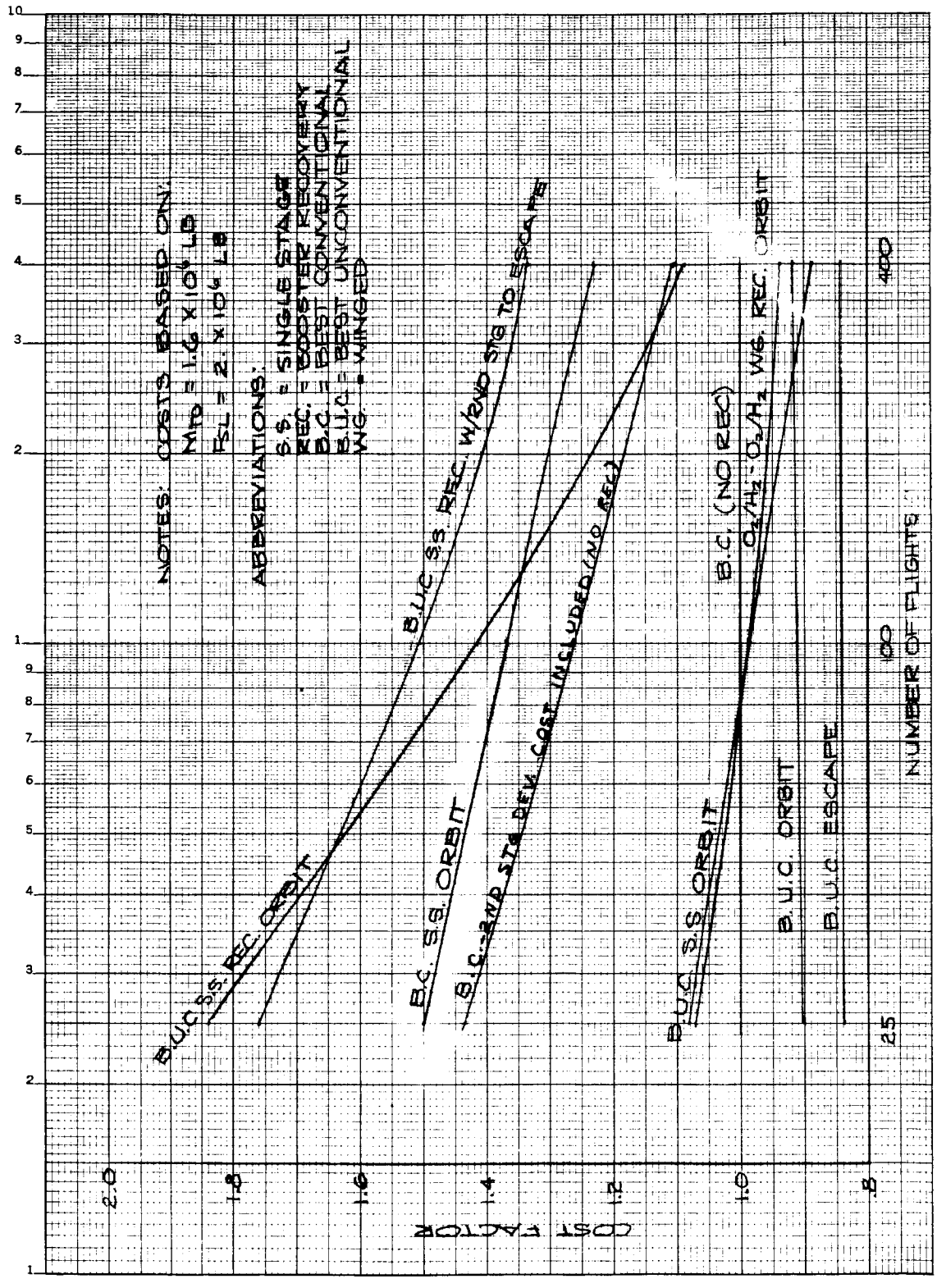


Figure 6

Comparison of Conventional vs Unconventional Configurations, Fixed Thrust

APPENDIX C

RECOVERY POTENTIAL FOR HIGH-VELOCITY BOOSTERS

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	1
II.	TECHNICAL DISCUSSION	2
	A. TECHNICAL FEASIBILITY	2
	B. RECOVERY CONCEPTS	10
III.	SUMMARY	14
REFERENCES		17

FIGURE LIST

	FIGURE NO.
Variation of Heating Rate During Re-entry	1
Total Heat Flow During Re-entry	2
Deceleration During Re-entry	3
Oscillatory Motion of a Ballistic Vehicle	4
Dependence of Wing Loading on Structural Temperatures for a Radiation-Cooled Vehicle	5
Excursion of Trimmed Aerodynamic Center with Mach Number	6

I. INTRODUCTION

This appendix presents a preliminary discussion of the technical problems and economic potential for recovery of first-stage rocket booster vehicles that achieve burnout conditions ranging from 15,000 ft/sec to circular orbit velocities and higher.

Numerous studies and conceptual designs for recovery of booster vehicles have been completed, invariably pertaining to vehicles with burnout velocities of less than 10,000 ft/sec. [1] These studies have typically concluded that recovery of expended liquid-propellant rocket boosters is technically feasible, and significant economic savings through the re-use of booster hardware is potentially available. These savings are based upon realistic estimates of recovery and rejuvenation cost factors; however, values for these factors differ considerably between sources, and the actual magnitude of low-velocity booster recovery savings will remain somewhat in question until pilot booster-recovery programs are seriously undertaken.

Preliminary conclusions of the current over-all booster vehicle study have indicated that orbital payload may be most economically delivered by rocket systems in which the first-stage booster delivers a large fraction of the total velocity requirement. Therefore, the booster vehicles may attain burnout velocities ranging from approximately 15,000 ft/sec, to 300 nm circular orbit velocity for a high-performance, single-stage-to-orbit version. Because achieving minimum total program cost-per-lb of delivered payload is one of the major objectives of concepts developed in this booster study, it will be desirable to investigate the feasibility and economic potential of recovery and reuse for these high-velocity boosters. Available booster-recovery studies have been limited to low-burnout velocities (less than 10,000 ft/sec); therefore, this discussion is limited to preliminary feasibility and

I, Introduction (cont.)

economic considerations, suggesting further and more detailed studies in this area.

II. TECHNICAL DISCUSSION

The logical inclusion of recovery operations in a booster program is predicted upon the following considerations:

Recovery techniques must be technically feasible and reliable, based on the basis of existing technology or technology that can be reasonably developed during the booster development period.

Recovery equipment must not excessively penalize the ability of the booster to satisfy its basic objectives.

Recovery and reuse of the booster hardware must result in significant economic savings for the total program, after all pertinent and chargeable costs have been considered.

A. TECHNICAL FEASIBILITY

1. The technical feasibility of recovery and reuse for high-velocity boosters can be definitely established only after a detailed investigation of all critical areas, including:

a. Re-entry heating
b. Re-entry aerodynamic loads
c. Impact loads
d. Capabilities of drag devices and lifting re-entry techniques
e. Stability and control
f. Booster-recovery system design compatibility

II, A, Technical Feasibility (cont.)

Preliminary considerations in these areas which indicate the nature of the technical problems involved are presented in the following sections.

2. Re-Entry Heating

The most critical problem involved in recovery of booster vehicles or vehicle components from orbital or near-orbital velocities appears to be the very high heating rates and total heat input to be expected during re-entry.

To illustrate the magnitude of heating encountered by a re-entering vehicle, Figure 1 pertains to the mean heating rate for the re-entry of a conical body of representative diameter at approximately 23,000 ft/sec. It is evident that a turbulent boundary-layer condition results in an appreciably greater heating rate--10 to 30 times greater than with a laminar boundry-layer condition.

For both types of boundary-layer conditions, the heating rate rises exponentially with time at the beginning of re-entry into the atmosphere. For bodies with low value of the drag/weight parameter, this exponential increase continues and a maximum heating rate occurs at low altitudes. For body with a large value of the drag/weight parameter, this exponential increase continues and a maximum heating rate occurs at low altitudes. A body with a large value of the drag/weight parameter suffers appreciable deceleration at high altitudes and consequently, has a lower heating rate at lower altitudes. For moderately high surface temperatures and a reasonably large value of the drag/weight parameter, the heating rate drops to zero and cooling actually occurs in the lower atmosphere.

The weight of a cooling system or heat shield is determined approximately by the total heat input during re-entry. Figure 2 shows the total heat input

II, A, Technical Feasibility (cont.)

per unit frontal area as a function of $(C_D A_c)W(\sin \theta)$ for laminar and turbulent boundary layer conditions. This figure pertains to the same re-entry conditions as for Figure 1. The advantages in maintaining a laminar boundary-layer condition and in using a design with a large value of $(C_D A_c)/(W \sin \theta)$ are evident.

A high-velocity booster vehicle, which is representative of the general type under consideration and considered as a standard, is the single-stage-to-orbit liquid oxygen/liquid hydrogen vehicle defined by the Aerojet-General booster investigation. The entire booster, with payload, achieves a 300 nm orbit prior to separation. This standard vehicle will be decelerated slightly prior to atmospheric re-entry, so that the curves in Figures 1 and 2, based on a 23,000 ft/sec entry velocity, are applicable. For a $(C_D A_c)(W \sin \theta)$ of 0.02, a mean total heat input of 4,000 Btu/ft^2 is inducated by Figure 2. This heat input for the standard vehicle is obviously intolerable for an unprotected aluminum skin because it predicts a temperature rise on the order of 5,000° F for a 0.25-in. thick section. If the $(C_D A_c)/(W \sin \theta)$ value could be increased by a factor of 20 through small re-entry angles and use of a large hypersonic decelerator, the predicted temperature rise would be reduced to approximately 1,300° F. This is still excessive for aluminum surfaces; however, the LH_2 tank will be insulated. By insulating the liquid oxygen - liquid hydrogen tank also with a material of appropriate characteristics, the heat input to the structure might possibly be held to tolerable levels.

A possible improvement in the heating problem might involve use of titanium or stainless steel tankage which can tolerate higher temperature without permanent effects upon material properties.

An additional alternative would involve recovery of only compact but expensive booster components for re-use. Using relatively large re-entry angles, these components could be easily and economically protected by ablative

II, A, Technical Feasibility (cont.)

heat shields while the more fragile components would either be separated before re-entry or would be destroyed by re-entry heating.

3. Re-Entry Aerodynamic Loads

During re-entry, the vehicle will be subjected simultaneously to large accelerations along the flight path and to accelerations caused by oscillations resulting from the initial misalignment.

Axial acceleration loads are illustrated in Figure 3 for re-entry velocity of approximately 23,000 ft/sec and an initial entry angle, θ, greater than 10° below horizontal. It may be observed that the altitude of maximum deceleration increases as the drag/weight parameter increases. Typical decelerations are on the order of 50 g; however, by the use of smaller entry angles, the maximum deceleration can be reduced as follows:

θ (°)	Maximum Deceleration (g)
1.5	8.5
3	11
5	16

Thus, decelerations can be held within quite acceptable limits if the use of small re-entry angles is acceptable from other standpoints (such as heating and range dispersion).

The oscillations resulting from the initial misalignment are illustrated in Figure 4. Besides subjecting the re-entry to high lateral loads, oscillations affect the distribution of aerodynamic heating. If only local heat

II, A, Technical Feasibility (cont.)

shielding on frontal areas is used, the re-entering vehicle must be sufficiently stable to reorient itself within about ± 4° of the line of flight by the time high heating rates are reached to avoid an excessive penalty in heat-shield weight.

4. Impact Loads

The most sophisticated recovery techniques for integral booster vehicles, such as winged systems or vertical-landing systems with rocket-powered hovering capability, will effectively eliminate impact loadings. However, simple techniquesusing only aerodynamic drag devices and possible small retro-rockets for terminal decelation will definitely result in impact leadings of various magnitudes.

If it is possible to program recovery over water, and immersion does not damage reusable components, a water impact is preferred to a land impact because of the lower impact deceleration. Decelerations caused by high-velocity water impact with nonstandard shapes cannot be accurately calculated at this time; the information that is available is empirical in nature. Initial impact produces a high deceleration for a very short period which is usually absorbed in the deflection of the contacting portion of the structure. This pulse is followed by a much lower peak as the water flow pattern is established, and this peak decays rapidly as the velocity decreases. Experiments have demonstrated that, by encasing objects in a slightly elastic medium such as plastic foam, the deceleration peak shows a marked decline at points removed from the outer shell. Hence, filling voids in impacting sections with a foamed-in-place plastic provides advantages both in shock attenuation and in ensuring flotation if leaks or damage should occur.

II, A, Technical Feasibility (cont.)

5. Drag Devices and Lifting Re-Entry

Use of conventional drag devices for deceleration prior to re-entry is limited by both heating and stability problems. The high stagnation temperatures would destroy lightweight fabric drag devices before they had achieved significant deceleration. Parachute-type devices are also found to be unstable at hypersonic speed; the use of inflated balloons or rigid conical shapes is necessary. These considerations result in weight penalties for hypersonic decelerators which are usually excessively high for use in booster recovery, particularly for near-orbital burnout velocities.

At a given velocity and for a given re-entry vehicle, critical heating rates are reduced by increasing the lift coefficient, or by flying at higher altitudes. This may be observed from the following relationship between heating rate, altitude, and lift coefficient in equilibrium flight

$$q \propto \left(\frac{W}{S}\right) \qquad \left(\frac{\left[1 - \left(\frac{V^2}{V_S{}^2}\right)^{1/2}\right] V_R{}^{3.15}}{e^{1/2}\,(C_L \cos \phi)\,1/2} \right) \propto \left(\frac{\left(e\right)^{-(h/28)}\, V_R{}^{3.15}}{e^{1/2}} \right)$$

A radiation-cooled surface reaches equilibrium at a temperature which balances the heat input with the heat radiation.

$$T_C \sim \left(\frac{q}{\epsilon}\right)^{1/4}$$

Thus, critical surface temperatures are also reduced with increasing lift coefficient. The temperature allowables of the materials and structures determine, to a large

II, A, Technical Feasibility (cont.)

extent, the wing loading (w/s)of the re-entry vehicle. Figure 5 illustrates this fact by a typical example of a wing with laminar heating of the stagnation point and turbulent heating at a certain point on the bottom surface.

At the hypersonic velocities, where the re-entry vehicle will operate, methods are currently available to calculate pressure and heat transfer distributions over relatively simple configurations with confidence. Examples are blunt bodies of revolution, flat plates at angle of attack, leading edges with sweep, etc. On more complex shapes, and particularly on problems of a local nature, semiempirical methods may be used where necessary, supported by extensive experimental data. Such situations arise in regions of interfering boundary layers, shock impingement on a surface, regions of flow separation and reattachment, etc., such as may exist over control surfaces, wing-body or fin-wing junctions, and local problems around holes, cracks, surfaces roughness, and discontinuities.

Many other considerations such as wing sweep, landing performance, stability and control, structural requirements, etc., must be considered to complete an evaluation of lifting re-entry for booster vehicles. Further studies in this area are definitely suggested in connection with high-velocity booster recovery; however, because the use of aerodynamic lift appears to be a feasible method for reducing the critical heating problem in recovery of the intact booster vehicle from near-orbital velocities.

6. Stability and Control

The flight spectrum of the re-entry vehicle covers the ranges of velocities from low subsonic through transonic, supersonic, hypersonic up to orbital velocities. The altitudes range from sealevel to several hundred miles above the

II, A, Technical Feasibility (cont.)

earth surface. Over this wide spectra of Mach numbers, Reynolds numbers, dynamic pressures, angle of attack, the re-entry vehicle must be stable, and for lifting vehicles, fully controllable.

In flying through the hypersonic down to subsonic velocities, a substantial excursion takes place in the location of the aerodynamic center travel. In the hypersonic flight regime and at hypersonic flight attitudes, the aerodynamic center shifts slowly to the rear with increasing Mach number, while in the transonic range, it moves far forward. This variation will ofter result in critical stability problems, either for ballistic or for lifting re-entry trajectories.

7. Booster-Recovery System Design Compatibility

The most effective and practical recovery concepts normally require complete integration with the vehicle system. The recovery system cannot be "tacked on" after the vehicle is made optimum for its basic mission; rather, the concept of recovery must be carried throughout the system design to achieve the most economical over-all operation. Thus, the concept of booster recovery should be integrated into the vehicle design and development program as soon as possible; preferably when the over-all booster program is still in its conceptual stages.

8. Deorbiting Propulsion

For a single-stage-to-orbit vehicle, it will be necessary to provide retrothrust in orbit to initiate re-entry. If the vehicle is to be recovered intact, the main engine may be restarted to provide the necessary velocity increment. However, if only compact but expensive components are to be separated and recovered, it may be more economical to separate the package in orbit and deorbit using solid-propellant retro-rocket. Figure 7 presents a curve indicating the retro-rocket

II, A, Technical Feasibility (cont.)

weight requirements to deorbit with various re-entry angles from a 300 nm orbit.

For the representative single-stage-to-orbit vehicle discussed above, the difference in I_{sp} between the main engine and solid-propellant retro-rockets results in a roughly equivalent weight penalty for either deorbiting the entire vehicle with the main engine, or separating the engine section (with guidance components attached) and deorbiting it with auxiliary solid rockets. The approximate propellant weight requirements for either case are indicated below:

Entry Angle (°)	Propellant Weight (lb)*
0	2,300
5	6,200
10	16,000

It is evident that achieving moderately high entry angles would require loss of a significant fraction of the vehicle's orbital payload capability.

B. RECOVERY CONCEPTS

Considering the requirements and limitations imposed by the technical feasibility discussion, three basic recovery concepts are suggested for further consideration. The very preliminary feasibility work included here can in no way guarantee the complete technical feasibility or practicality of these concepts. However, it does tend to indicate that they are each basically feasible, assuming the success of necessary but reasonably straight-forward development efforts in some areas.

---

* On basis of 90,500-lb intact vehicle weight or 45,000-lb recoverable engine-guidance package.

II, B, Recovery Concepts (cont.)

1. Component Package Recovery

The recovery of a package made up of compact but expensive booster components is one of the alternatives. This package could be protected by a relatively small amount of ablative material during re-entry; no control systems or auxiliary aerodynamic decelerators would be used during the re-entry period. It would then descend to water impact under conventional parachutes, with subsequent retrieval by a surface vessel. In most cases, the package should probably be separated from the remainder of the booster vehicle (tankage and airframe) prior to re-entry, to guarantee predictable flight characteristics during re-entry and terminal descent. However, in some instances it may be possible to use the remainder of the vehicle to provide beneficial drag, heat-shielding, and impact absorption functions, even though it is destroyed or irreparable damaged during re-entry.

A review of the characteristics of the representative single-stage-to-orbit vehicle indicates that it is well-suited for component recovery. With minor design modifications, the component package for this vehicle might logically include the entire forced-deflection engine and pump assembly, and all possible on-board components of the guidance, programer, and telemetry gear. The heavy turbopump would cause the engine section to re-enter in stable flight with the pump forward. All expensive electronic components that could be transferred to the turbopump area would be included in the package*; an ablative heat shield would be

* It is realized that some of the guidance and telemetry components could not be located in this area because of considerations such as vibration, heating, location relative to the c.g., etc.

II, B, Recovery Concepts (cont.)

added forward of the pump to provide protection during re-entry. Because most of the components included can tolerate high g loads, the package could re-enter with a relatively steep entry angle, resulting in short heating durations and efficient use of ablative heat-protection materials.

The package would be equipped with conventional parachutes for descent after re-entry, and would easily withstand the associated water impact at landing. With proper design of the best shield and package, the unit could be rendered naturally bouyant, and would subsequently be picked up by a surface ship for return to the rejuvenation and check-out facility.

2. Intact Recovery, Drag Only

The second concept suggested for further consideration involves recovery of the entire vehicle using an aerodynamic drag device to provide deceleration at high altitudes, resulting in reduced aerodynamic heating. This concept must be predicated upon development of a suitable hypersonic decelerator for this re-entry application; possibilities include the use of flexible stainless steel mesh fabrics in collapsible drag devices, or the incorporation of a rigid petal-type decelerator which forms part of the vehicle skin during ascent. The decelerator is certain to be relatively heavy, resulting in a considerable loss of booster payload.

The entire vehicle would first be deorbited, probably by use of the main propulsion system. It would then follow a very flat re-entry trajectory, with the decelerator deployed for stability as soon as adequate dynamic pressure was encountered. By using a flat re-entry, the g-loadings on the vehicle and the forces on the drag device would be held down, and the decelerator would be used in the most efficient manner. Although maximum heating rates are reduced in

II, B, Recovery Concepts (cont.)

this type of re-entry, stagnation temperatures will be very high, requiring high-temperature decelerator materials and possibly insulation of the vehicle structure. After re-entry, the vehicle could be lowered by conventional parachutes for either water or land recovery, with auxiliary retrorockets used to achieve a softlanding or possibly even provide some hovering capability for a controlled landing.

Application of this recovery technique to the representative single-stage-to-orbit vehicle has been considered, indicating several distinct advantages and some major problems. First, the single-stage-to-orbit concept allows one complete orbit prior to recovery, which can return the vehicle reasonably near the original launch site. The existing insulation of the LH_2 tank will be of considerable value in limiting the wall temperature during this type of re-entry; insulation would also probably be added for the LOX tank and other exposed vehicle skin. The considerable residual cryogenic propellant aboard the vehicle (almost 15% of the burnout weight) might also be considered as a medium or heat sink for several conceivable cooling schemes. This residual propellant could maintain tank pressurization during re-entry to improve the vehicle's structural stability. Use of titanium or stainless steel rather than aluminum for tanks and vehicle structure would allow considerably higher skin temperatures during re-entry. The most predominant disadvantage noted was the large size and high weight of applicable hypersonic decelerators. Advances in fabrication materials and development of improved decelerator concepts will be necessary to implement this recovery concept in a practical manner.

3. Intact Recovery, With Lift

Many techniques and system concepts can be conceived for the use of lifting re-entry and subsequent lifting flight in high-velocity booster recovery. This section will attempt only to point out a few of the basic characteristics and merits of such operations.

II, B, Recovery Concepts (cont.)

As noted in the technical feasibility section, the use of lifting trajectories can considerably reduce critical heating rates during re-entry. The controllable lifting vehicle can also be made to glide to the desired landing area, its maneuvering capabilities dependent primarily upon the L/D ratio obtained.

The amount of lift achieved and the method by which it is obtained can vary widely between concepts. For example, the use of hypersonic wings and control surfaces could effectively transform the expended booster into a controllable re-entry glider. On the other hand, a significant amount of lift can be obtained at the higher velocities by using a reaction-jet attitude control system to maintain the proper vehicle attitude relative to the flight path. Intermediate capabilities may be obtained by modification of the vehicle's basic shape or use of concepts such as inflatable or Rogallo wings at lower velocities. In general, the practical application of lifting re-entry techniques will require significant changes in the booster design concepts and will result in considerable increases (50 to 200%) in the vehicle's inert weight, with the associated loss in booster payload capability.

Lifting re-entry for the representative single-stage-to-orbit vehicle would involve considerable modification of its basic design concepts. However, the use of lifting recovery is not to be entirely discounted at this time, and should receive further consideration in any study of recovery techniques for this vehicle.

III. SUMMARY

In summary of the preliminary considerations presented in this discussion, the following major points are evident.

III, Summary (cont.)

A. Re-entry heating is the most formidable problem in recovery of boosters with burnout velocities ranging from 15,000 ft/sec to orbital speeds and above.

B. The most straightforward recovery concept for high-velocity boosters from a re-entry heating standpoint is the recovery of packages containing only compact but expensive components.

C. Concepts for intact vehicle recovery based on development of a practical hypersonic decelerator, and use of lifting re-entry trajectories, show potential feasibility and should be considered in further high-velocity booster recovery studies.

D. If anticipated savings are felt to be adequate to justify serious consideration of recovery operations in a conceptual launch-vehicle program, basic studies of recovery techniques and design requirements should be carried out concurrently with conceptual vehicle design work. This approach is absolutely necessary to guarantee an integrated system design providing maximum economic returns.

SYMBOLS LIST

q	=	Heat flow
w/s	=	Wing loading
C_L	=	Lift coefficient
$\ell(1/2)$	=	Characteristic length
V	=	Inertial vehicle velocity
V_S	=	Satellite velocity
V_R	=	Free-stream velocity
ϕ	=	Bank angle
h	=	Altitude
β	=	Constant in atmospheric component
T_C	=	Radiation equilibrium temperature
ϵ	=	Emissivity

REFERENCES

1. Aerojet-General Corporation, Booster Recovery Study, Report No. 1816, July 1960

2. Allen, H. Julian, Motion of a Ballistic Missile Angularly Misaligned With the Flight Path Upon Entering the Atmosphere and its Effect upon Aerodynamic Heating, Aerodynamic Loads, and Miss Distance, NACA TN 4048, October 1957

3. Bulban, E. J., Rotor Recovery Proposed for Boosters, Aviation Week, p. p. 115, 18 April 1960

4. Detna, R. W., Riddell, F. R., and Rose, P. H., Controlled Recovery of Non-Lifting Satellites, ARS Paper 784-59, May 1959

5. Haig, C. R. Jr., The Use of Rotors for the Landing and Re-Entry Braking of Manned Spacecraft, IAS Paper No. 60-17, 1960

6. Jones, D. J. and Tearnen, J. O., Winged Recoverable Booster Characteristics, ARS Paper No. 1013-59, 1959

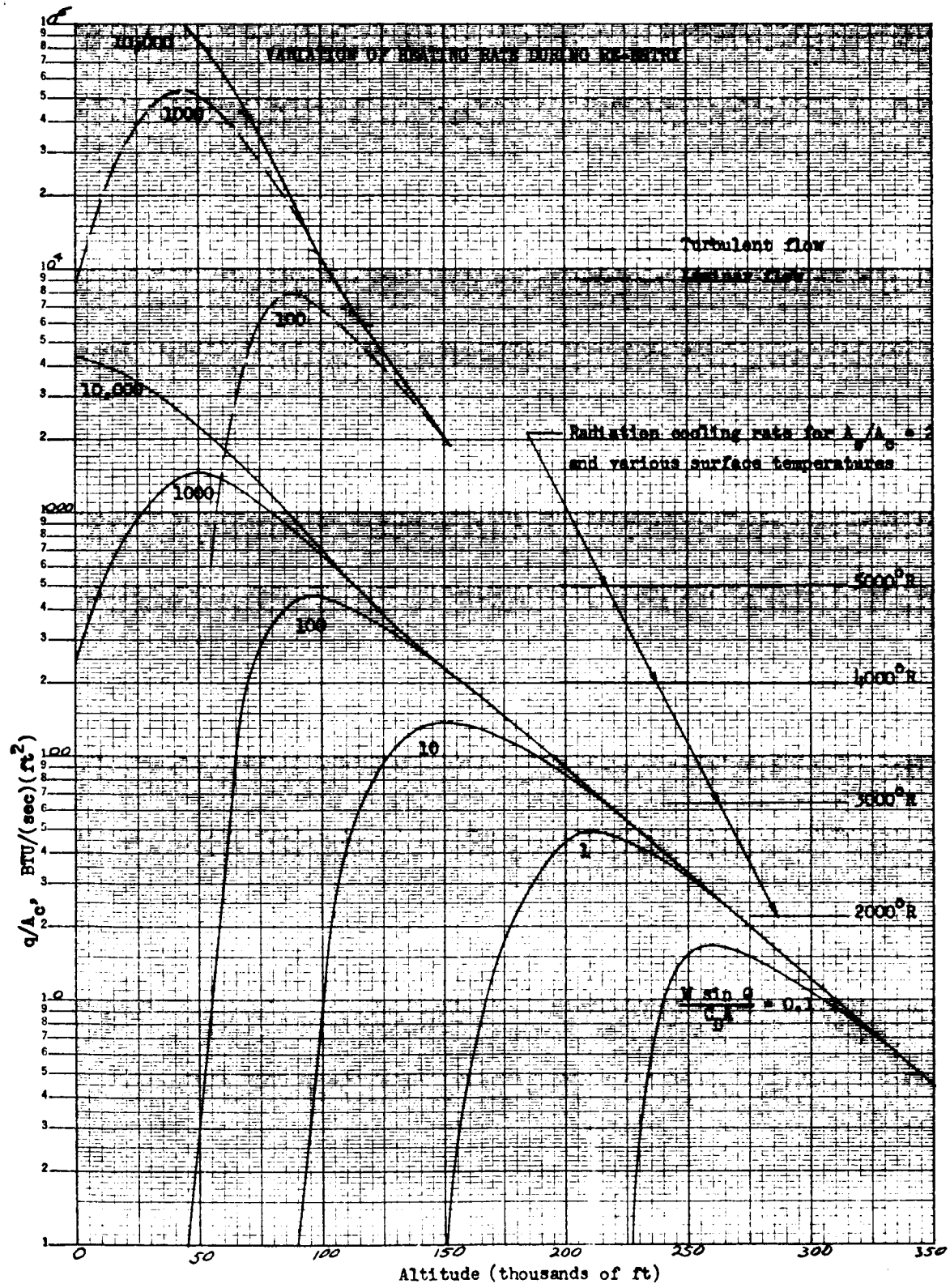
7. Mazelsky, B., The Application of Rotary Wings for Recovery of Space Vehicles, Aerojet-General Report TR:D:1, January 1960

8. McDonnell Aircraft Corp., Recoverable Booster Support Systems Study, Final Summary Report, Report 6996, ARDC-TR-46, September 1959 (Secret)

9. Romick, D. C., Belfiglio, R. A., and Sandgren, F. B., "Recoverable Boosters are Studied to Cut Manned Space Flight Cost," Missions and Rockets, p. 95-97 (April 1958)

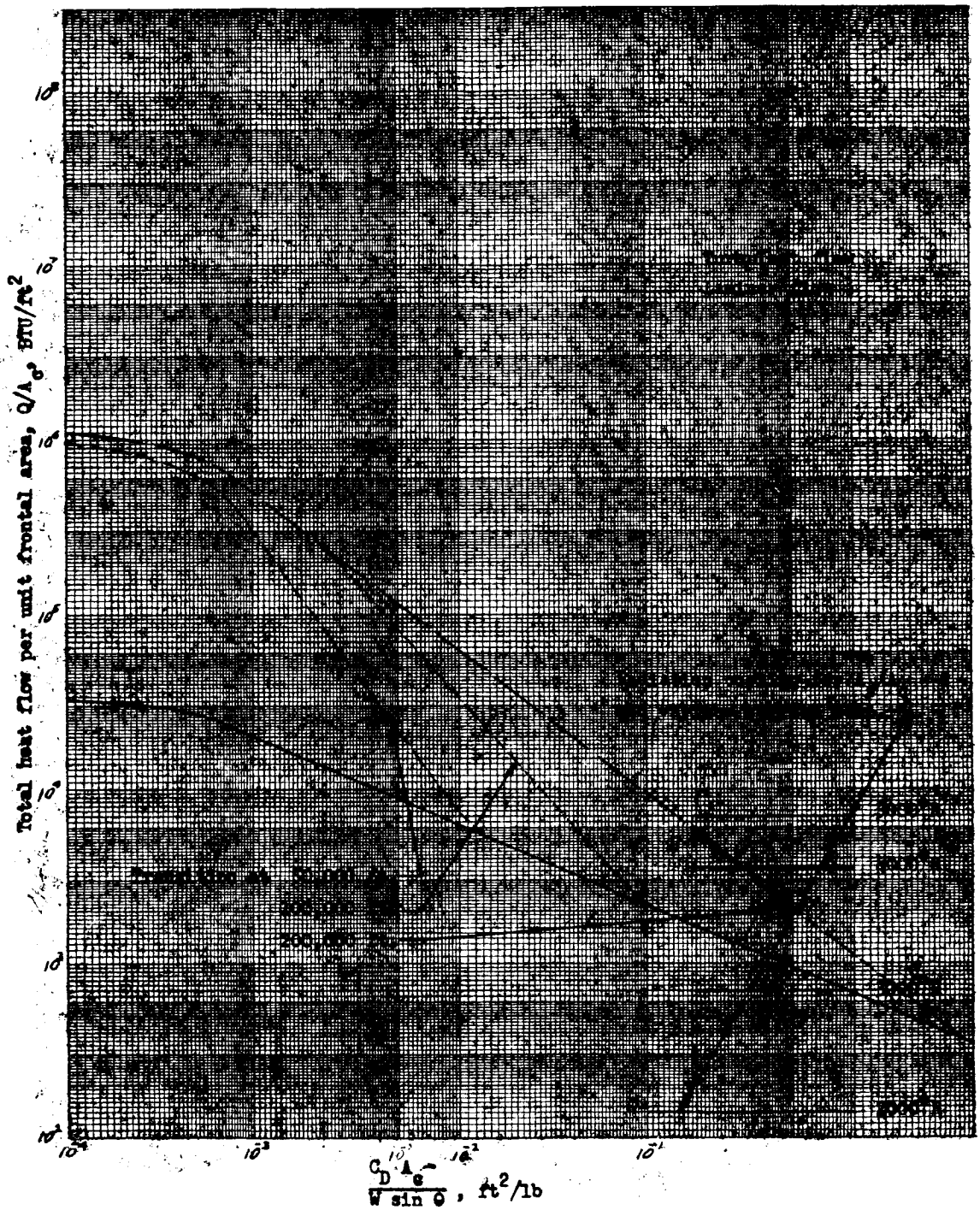
10. Shipps, P. R., On the Methods and Economics of Recovering Boosters for Orbital Vehicles, IAS Paper No. 60-29, 1960

(End text)



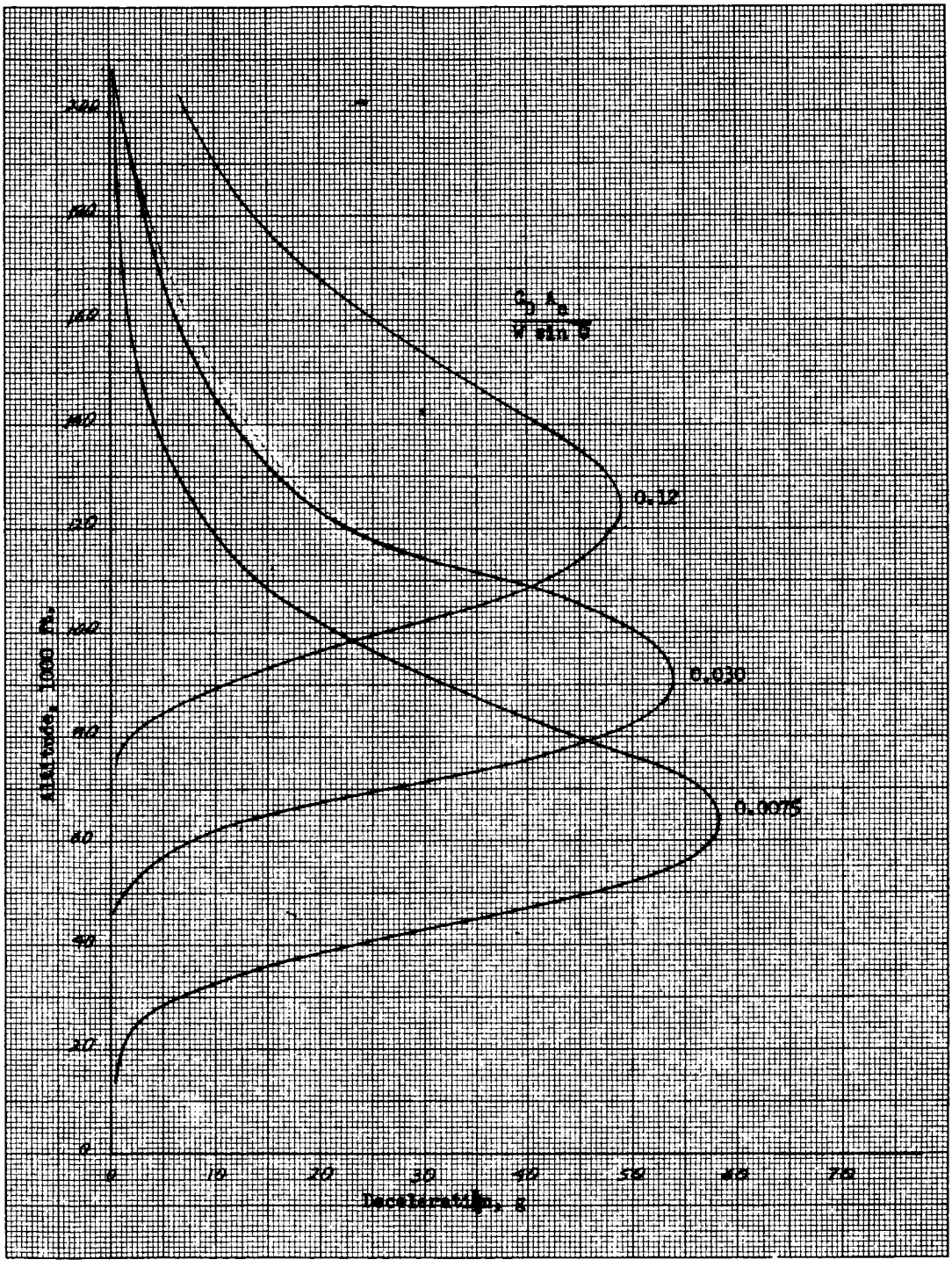
Variation of Heating Rate During Re-entry

Figure 1



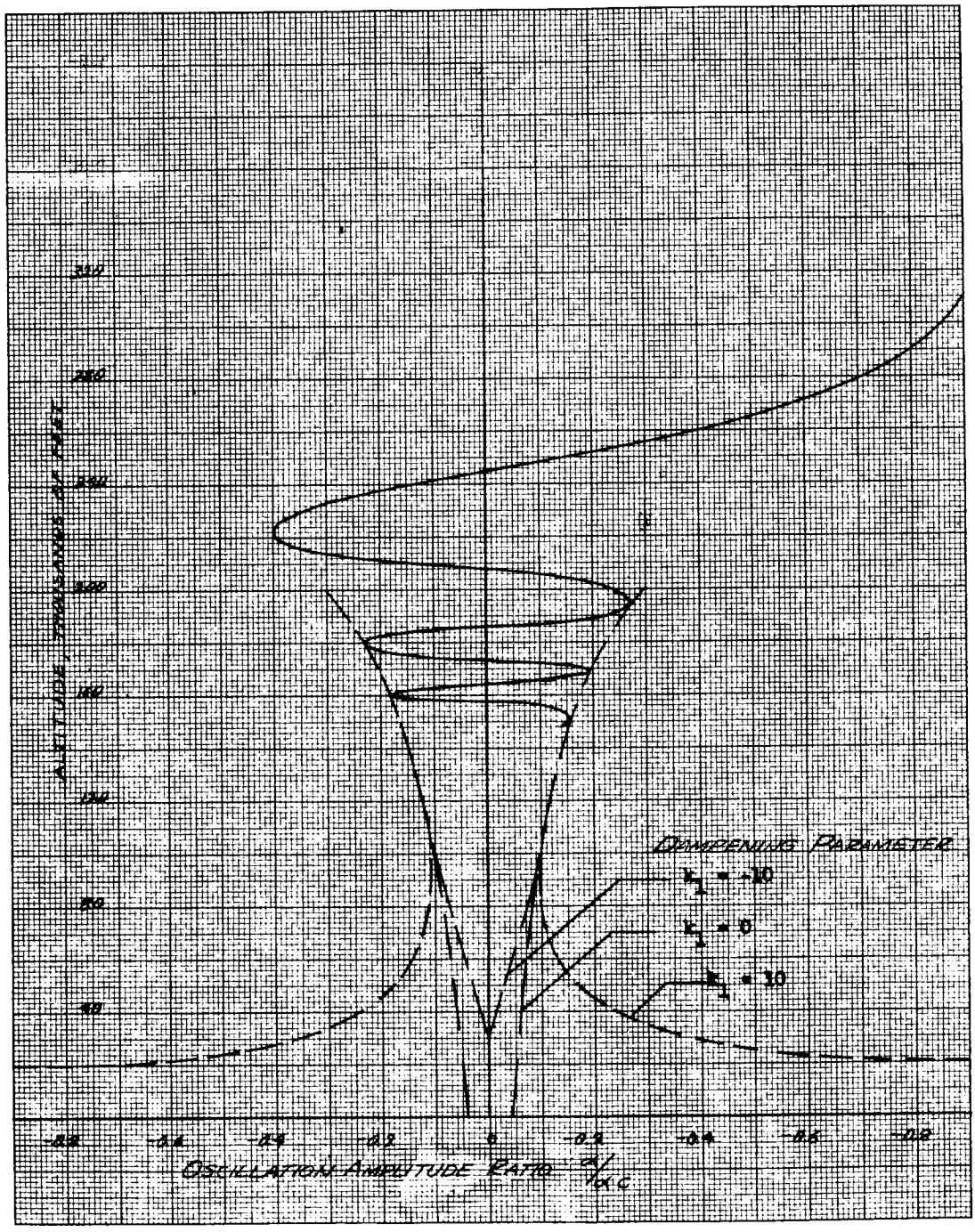
Total Heat Flow During Re-entry

Figure 2



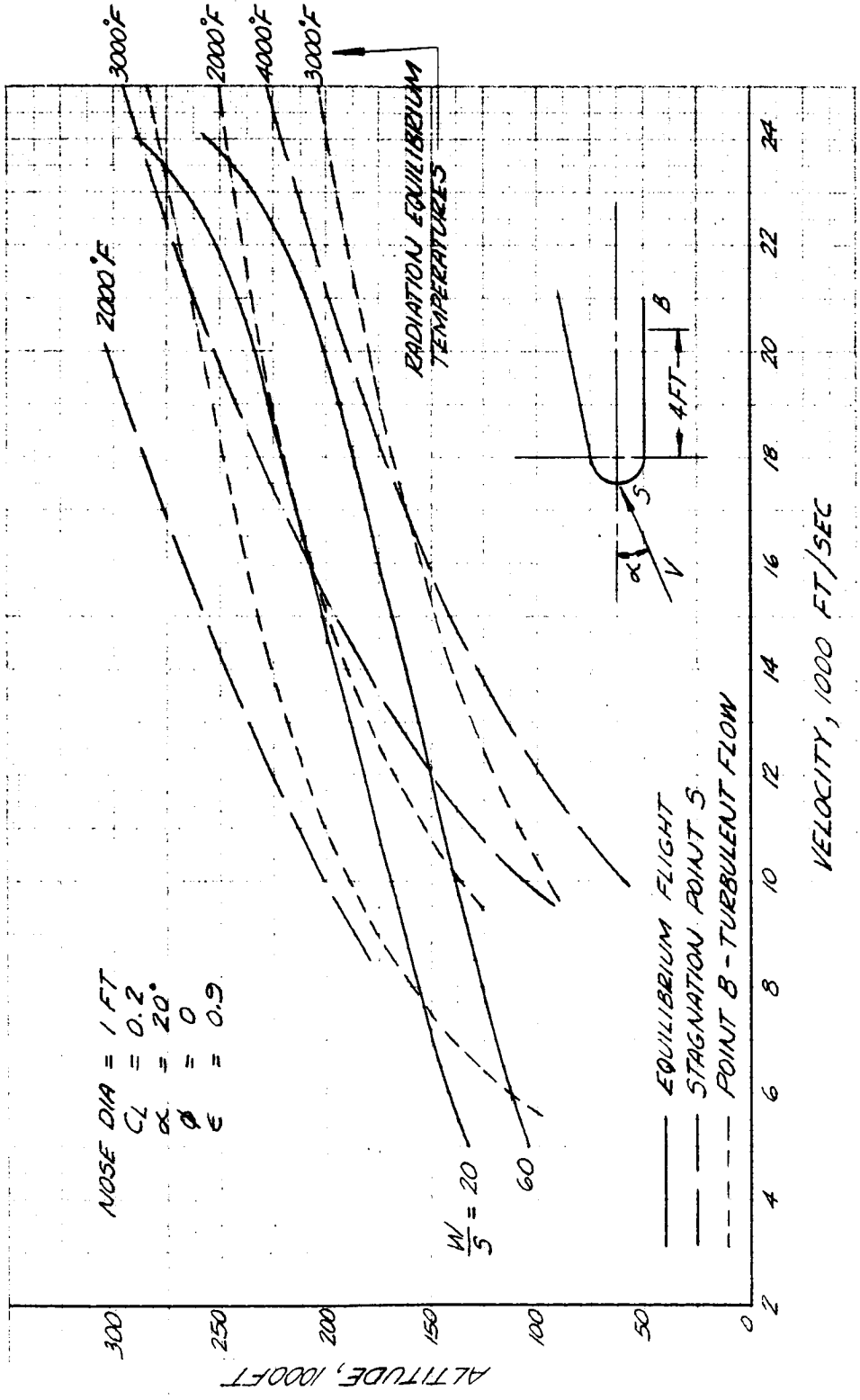
Deceleration During Re-entry

Figure 3



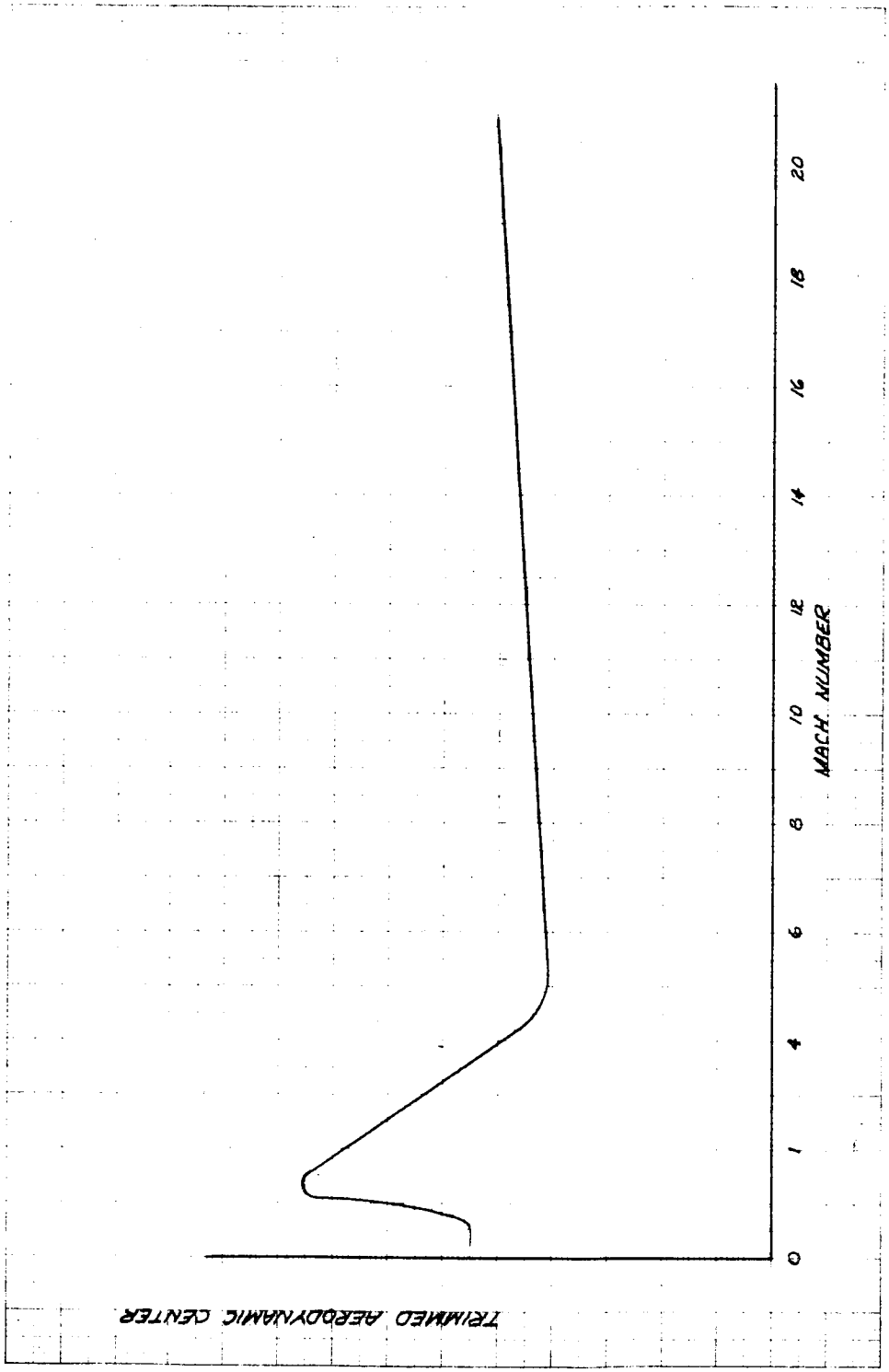
Oscillatory Motion of a Ballistic Vehicle

Figure 4



Dependence of Wing Loading on Structural Temperatures For a Radiation Cooled Vehicle

Figure 5



Excursion of Trimmed Aerodynamic Center with Mach Number

Figure 6

APPENDIX D

HIGH-THRUST-PER-INJECTOR ELEMENT

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	I-1
II.	SUMMARY	II-1
	A. CONTRACT AF 33(616)-5220	II-1
	B. CONTRACT AF 04(647)-521	II-2
III.	TECHNICAL DISCUSSION	III-1
	A. GENERAL	III-1
	B. CONTRACT AF 33(616)-5220	III-6
	C. CONTRACT AF 04(647)-521	III-27
	D. VARIABLE THRUST APPLICATION	III-32

TABLE LIST

	TABLE NO.
Large Thrust Per Injection Element Performance	1
Large-Thrust-Per-Element Injectors used for Phase II (Reactive) Tests	2
Theoretical Performance of N_2O_4/Aerozine-50 and N_2O_4/UDMH	3
Large Thrust Per Element Injector Performance with N_2O_4/Aerozine-50	4
Test Data Large Thrust/Element Combustion Stability 22.5-in.-dia Pulse Motor, Propellants N_2O_4-Aerozine-50	5
Aerozine-50 Composition	6
Phase III Results of Large Thrust Per Element Injector Program	7

FIGURE LIST

	FIGURE NO.
Characteristic Exhaust Velocity vs Mixture Ratio for N_2O_4/Aerozine-50	1
Specific Impulse vs Mixture Ratio for N_2O_4/Aerozine-50	2
Thrust Coefficient vs Mixture Ratio for N_2O_4/Aerozine-50	3
Characteristic Exhaust Velocity vs Mixture Ratio for N_2O_4/UDMH	4
Specific Impulse vs Mixture Ratio for N_2O_4/UDMH	5
Thrust Coefficient vs Mixture Ratio for N_2O_4/UDMH	6
Large-Thrust-per-Element Injector Phase, Phase I	7
22.5-in. Dia Pulse Motor Installation	8
Injector Element Orientations and Stability Contours for Large-Thrust-per-Element Triplet Injector	9
Design and Spray Characteristics--AIT Mod 2F-8 Injector Element	10
Nonreactive Spray Test Installation	11
Sampling Probes	12
Mixture Ratio Sample Analysis Equipment	13
Orientation of Sample Probes	14
Large-Thrust Injection Element (5,000 lb)	15
Splash-Plate Element--AIT Mod 1	16
Splash-Plate Spray Pattern--AIT Mod 1	17
Flow-Rate and Mixture-Ratio Distribution AIT Mod 1 (Splash Plate) Element	18
Triplet Element, AIT Mod 2 Type	19
Typical Triplet Element Installation	20
Triplet Spray Pattern--AIT Mod 2	21
Flow-Rate and Mixture-Ratio Distributions for AIT Mod (Triplet) Element (ϕ = 45°)	22

FIGURE LIST (cont.)

	FIGURE NO.
Flow-Rate and Mixture-Ratio Distributions for AIT Mod 2A and 2B (Triplet) Elements (ϕ = 60° and 35°)	23
Large-Thrust Transparent-Chamber Test-Installation Schematic	24
Transparent Thrust Chamber	25
Triplet Spray Pattern AIT Mod 2C	26
Design and Spray Characteristics AIT Mod 2C Injector Element	27
Design and Spray Characteristics AIT Mod 2C-1 Injector Element	28
Percent of Theoretical Characteristic Exhaust Velocity vs Momentum Ratio for N_2O_4/Aerozine-50	29
Design and Spray Characteristics AIT Mod 2C-1A Injector Element	30
Comparison of Triplet Elements	31
Theoretical Characteristic Velocity vs P_c for N_2O_4/Aerozine-50	32
Specific Impulse vs Chamber Pressure for N_2O_4/Aerozine-50	33
Thrust Coefficient vs Chamber Pressure for N_2O_4/Aerozine-50	34
c* vs Mixture Ratio for N_2O_4/Aerozine-50 Large Thrust per Element Injector Program	35
I_{sp} vs Mixture Ratio for N_2O_4/Aerozine-50 Large Thrust per Element Injector Program	36
Eroded Injector Element	37
Spray Patterns of 3 POD LT/E Injector	38
Comparison of 1-Element Injector to 3-Element Injector Performance (L* = 51)	39
Comparison of 1-Element Injector to 3-Element Injector Performance (L* = 59)	40
General Configuration of Triplet Injector Assembly (Push-Pull Cable)	41
General Configuration of Triplet Injector Assembly (Gear Train)	42
Water Spray Characteristics of 1,000 to 10,000 lb Thrust Variable Flow Injector	43

I. INTRODUCTION

The expanding space exploration program has placed demands on the rocket industry for engines with thrust levels far in excess of any previously developed engines. Increased thrust levels present many challenging problems, especially when the common engineering practice of scaling up existing designs is employed. One of the most complicated components to scale-up successfully with a limited amount of development time is the injection system. An injector designed along the conventional lines would require drilling many orifices which is time consuming and expensive. It would also require a complicated manifolding system to provide adequate injector face cooling which adds to the design time and fabrication expense.

In an attempt to improve the state-of-the-art for large thrust per injector applications, a program was initiated to investigate single element injectors. An element is defined as the smallest possible grouping of orifices that are dependent upon each other for efficient combustion, yet are relatively independent of other orifices. Preliminary work was undertaken to determine if efficient element designs in the range of 5,000 to 10,000 lb were possible and if even higher thrust per element designs were feasible. The design requirements stipulated that the injector element be simple, compact, and relatively self sufficient with respect to propellant supply, mixing, and burning, so as to realize high combustion efficiencies without the necessity of interaction with other injector elements.

Although this concept did not originate with Aerojet, significant accomplishments were made initially in work under Contract AF 33(616)-5220 and later continuing under Contract AF 04(647)-521, and have resulted in major advances in the state-of-the-art with regard to the simple, inexpensive designing and fabrication of large thrust injectors.

II. SUMMARY

A. CONTRACT AF 33(616)-5220

A two-phase test program was conducted to develop a large thrust-per-element injector. Phase I consisted of nonreactive liquid spray studies which established the spatial distribution of propellants and the mixture ratio distribution of selected large thrust injector designs. Phase II consisted of reactive tests to determine the combustion characteristics and to obtain performance data of an injector design selected on the basis of the Phase I results.

During Phase I, various methods of achieving good propellant mixing were studied. The results of these tests indicated that the most practical means of achieving atomization in a large-thrust element is by impinging the propellant streams, as in a triplet or splashplate element.

The spray patterns of several types of triplet and splashplate injectors were analyzed for spatial distribution and mixture ratio data using an acidic solution for the fuel and a basic solution for the oxidizer. From the results of these tests, a triplet element with a 90° impingement angle was selected for the Phase II tests.

Phase II of the program consisted of a study of the combustion characteristics of various triplet types of injector elements using the propellants N_2O_4/UDMH and N_2O_4/Aerozine-50*. The injectors were tested in a transparent-walled thrust chamber having a rectangular cross section. Highspeed motion pictures of the combustion tests were taken through a Plexiglas window.

* 50 wt % UDMH, 50 wt % N_2H_4

II, A, Contract AF 33(616)-5220 (cont.)

Two tests were conducted using N_2O_4/UDMH and eleven (11) tests were made using N_2O_4/Aerozine-50; all were satisfactory. The results of these tests are presented in Figure 1 through 6 and in Table 1. Steady-state experimental values of up to 97.8% of theoretical c* (based on shifting equilibrium) were achieved with N_2O_4/Aerozine-50 at optimum mixture ratio, at 500 psia chamber pressure, and at thrust levels above 13,000 lb.

The combustion tests conducted with the triplet injector, although limited in number, have definitely established the potential application of large-thrust injection elements to multimillion-pound thrust chambers. The development of the large-thrust element should result in a major contribution to the state-of-the-art.

B. CONTRACT AF 04(647)-521

1. General

A three-phase program was conducted to satisfy the following objectives: Phase I - obtain firm combustion performance and specific impulse data with a single-element injector; Phase II - determine the combustion stability characteristics of large-thrust elements; and Phase III - to compare multiple-element performance with the single-element performance. The injectors developed in this program were triplet-type elements with two oxidizer streams impinging on one fuel stream.

II, B, Contract AF 04(647)-521 (cont.)

Phase I consisted of ten tests using an uncooled single-element injector and a cylindrical uncooled thrust chamber. The injector was a triplet type with two elliptical-shaped oxidizer tubes and one square-shaped fuel tube. High performance was achieved with this design in tests conducted on Contract AF 33(616)-5220.

A portion of the test installation is shown in Figure 7. The propellant flow is controlled by two bipropellant valves; these valves are sequenced for a 50 millisec oxidizer lead. Also in Figure 7 is a view of the copper nozzle and the triple-element injector.

Performance was excellent with c* values up to 5930 ft/sec and I_{sp} values up to 94.5% of theoretical. Smooth hypergolic ignitions were obtained and stable steady-state operation was achieved with chamber pressure oscillations less than $\pm$ 3.5%.

Phase II consisted of 23 tests utilizing a 22.5 in.-dia pulse motor containing 12 large-thrust-per-element-type injectors. Each element consisted of two cylindrical-shaped oxidizer tubes and one rectangular-shaped fuel tube. These elements were located around the periphery of the injector face (Figure 8).

Tangential combustion instability was artificially induced by means of three pulse generators located on the outer circumference of the thrust chamber using standard pulse motor techniques. These generators introduced gas pulses of increasing intensity and were fired in sequence about 200 millisec apart.

II, B, Contract AF 04(647)-521 (cont.)

This permitted the inherent stability characteristics of various injector operating conditions (e.g., mixture ratio and injection velocity), and orientations to be evaluated by comparing the minimum pulse charge necessary to induce sustained tangential combustion instability. An automatic shutdown device monitored the induced instability and would terminate a test if stability exceeded the following criteria:

a.	pressure amplitude (peak-to-peak)	160 psi
b.	frequency	1,000 cps
c.	duration of a and b	40 millisec

The test program investigated two injector element orientations; (1) spray fan parallel to the chamber wall, and (2) spray fan perpendicular to the chamber wall. The positions are shown in Figures 9 (a) and 9 (c).

Good c* performance was achieved for both orientations. However, slightly better stability characteristics were exhibited with the spray fan perpendicular to the chamber wall.

Phase III consisted of ten tests conducted with an uncooled three-element injector of 3,500-lb-thrust-per-element and two uncooled cylindrical thrust chambers of different characteristic length. Performance was excellent with c* values from 95.9 to 100% of theoretical and I_{sp} values from 90 to 97% of theoretical. Smooth hypergolic ignition was achieved and stable steady-state operation was obtained with chamber pressure oscillations less than ± 4.5% of chamber pressure in all tests. Three-element injector performance is much higher than one-element injector performance when utilized in a small combustion chamber (L* less than 60 in.). Only minor performance and stability changes were noted for different element orientations.

II, B, Contract AF 04(647)-521 (cont.)

2. Titan Application

A program was initiated in February 1961, to adapt the large-thrust-per-element concept to the Titan II, XLR91-AJ-5 thrust chamber. The program was divided into two phases. Phase I concerned itself with subscale testing at the 10,000-lb-thrust level to obtain preliminary performance data on impinging sheet injector patterns. This evaluation was considered essential before designing the full-scale injectors because little was known about the characteristics of impinging sheet injectors. Phase II involves design, fabrication, and testing of full-scale XLR91-AJ-5 injectors based on the information obtained from the Phase I studies.

This program will also include a task to evaluate two unknown areas inherent in long-duration large-thrust-per-element injector testing. These two areas are: (a) the effect of injector heat "soak back" on the injection process and therefore, the combustion performance, and (b) the magnitude and location of chamber wall "hot spots" associated with single-element triplet injectors. This evaluation is presently being conducted and results are not yet available.

III. TECHNICAL DISCUSSION

A. GENERAL

A program was conducted to develop a single injector element having a thrust-level of up to 10,000 lb. Such an element should simplify the design and fabrication of injectors for large-thrust chambers by reducing the number of elements required.

In order to scale up a basic injection element, it is desirable to understand more fully the factors affecting propellant mixing and liquid atomization. Propellant mixing to produce a homogenous mixture is dependent on the degree of atomization of the propellants, contact time, and the interaction between the sprays from each injector element. Basic factors influencing the atomization of a liquid stream are: (1) aerodynamic drag, (2) other external forces, (3) liquid turbulence, and (4) surface-tension effects. Types of injection elements which involve combinations of aerodynamic drag and surface-tension effects (such as a showerhead element) do not appear suited for scaling up. The flow rate through such an element increases proportionately to the second power of the diameter, whereas the stream tube surface increases only as the first power of the diameter. Therefore, the mixing effect due to the aerodynamic drag or surface tension lessens as the element increases in size.

Injection elements in which atomization is governed primarily by liquid turbulence (such as hollow-cone elements) are also unsuitable for scaling-up for similar reasons. This leaves, as a possible method of achieving atomization in a large element, liquid disintegration by means of the application of some external force. The external force can be applied by the impingement of the streams, such as in doublet, triplet, or splashplate elements. In each of these types of elements, the momentum of the liquid stream striking against another stream or a stationary surface, results in the necessary external force application.

III, A, General (cont.)

Not only is it necessary to break up the injected liquids, but a high degree of mixing must also be achieved to prepare the propellant for rapid vaporization. This may be accomplished by impinging liquid-fuel and oxidizer streams upon each other.

The importance of the vaporization process in rocket engine combustion has been shown in previous experimental studies. These studies have shown qualitatively that atomization of the slower vaporizing propellant gave the greatest increase in combustion efficiency. Analytical studies at NACA (Reference 2), based on the concept of propellant vaporization as the rate-controlling combustion process, has shown how changes in drop size, gas velocity, drop velocities, chamber pressure, and propellant temperature affect performance. For instance, the vaporizing-rate calculations for Aerozine-50 and liquid hydrogen indicate that there would be large differences in combustion efficiencies if the same injector design criteria were used. However, specific studies of each of these variables under controlled test conditions are required to evaluate the concept. Such an evaluation is usually not warranted for an injector feasibility or development program because of the time and cost involved.

Stable combustion requires droplet or molecular contact between fuel and oxidizer in order to initiate reaction. This reaction forms the combustion products and releases heat. The heat further accelerates the vaporization of fuel and oxidizer and increases the rate of chemical reaction. Injectors, therefore, must be designed to cause formation and intimate mixing of droplets, resulting in rapid reaction and combustion.

III, A, General (cont.)

1. Impingement Angles

One of the design parameters to be considered is the impingement angle of the liquid streams. Most impinging stream designs produce a "fan" of droplets having a shape that is characteristic of the injector design. The droplet size and spray distribution in the fan is affected by the angle of impact and the distance of the impingement point to the face of the injector. Wide impingement angles tend to concentrate the fan nearer the injector face, whereas narrow angles result in the opposite effect. It has been considered that the maximum practical limit for spray angles is approximately 120°. It is possible, under some conditions to produce 130° spray angles, but these sprays have been subject to large magnitude perturbations in the fan, producing combustion instability. Extremely wide angles cause "drooling" particularly at lower injection pressure drops. For most designs with 60 to 90° angles, there is usually very little effect in atomization over a relatively wide range of differential pressures. A disadvantage of small impingement angles is that they produce a narrow fan angle which leaves void areas near the injector face. This causes extensive hot gas recirculation and subsequent erosion of the injector.

2. Injection Velocities (Momentum)

Injection velocities of impinging propellant streams are an important factor in obtaining fine atomization and intimate mixing. The significance of injection velocities is the tie-in with the momentum of the streams. The stream momentum, particularly the relative momentum, is a controlling parameter for determining both the droplet size and the droplet distribution resulting from impingement.

III, A, General (cont.)

The above criteria are important for any injector incorporating impinging streams. For a large number of small injection elements, the effect of stream momentum can become masked by the interaction between elements. Hence, in an injector having a closely spaced element matrix, intermixing between elements and hot gas recirculation can result in high combustion efficiency even though the atomization and mixing produced by one single element may be rather poor. As the size of the element is made larger however, surrounding elements have less effect and the individual characteristics of the single element becomes highly important. The importance of the momentum characteristics of a single injection element was demonstrated in test conducted under Contract AF 33(616)-5220. In those tests, it was found that the combustion efficiency obtained with a single triplet element varied from 78% to 95%, merely as a result of changing the relative momentum of the oxidizer and fuel streams.

For impinging cylindrical streams, it is important that each stream have a symmetrical turbulent flow velocity profile. An unsymmetrical velocity profile results in poor droplet breakup by impingement, that is, part of the fluid of the stream is moving at a different velocity than the remainder of the fluid in the stream and, therefore, will not produce the same impact forces upon impingement.

3. Effect of Stream Size

Some parameters will have a greater influence upon performance than others. The impingement angle and momentum ratio have been the most significant parameter with N_2O_4/Aerozine-50 triplet design. Here, the important parameter is the ratio of oxidizer to fuel orifice diameters and the impingement distant from the injector face. Secondly, the momentum level which is directly

III, A, General (cont.)

related to the orifice diamater is important since this momentum is the hydraulic force causing liquid breakup. Injector orifices (approximately 3/4 in.) have been successfully utilized, although the maximum level to which this parameter can be scaled-up is unknown.

4. Orifice L/D

Previous work conducted at Jet Propulsion Laboratory indicated that L/D ratios of 20 or more were necessary for propellant systems that permit liquid phase mixing to assure turbulent flow conditions at the exit. Studies at Aerojet-General Corporation have since shown the empirical data for N_2O_4/Aerozine-50 indicate that L/D ratios no larger than five are satisfactory for this concept. It is believed that this ratio can be further reduced providing velocities are in the range of the high Reynold's Numbers.

5. The Effect of P_c

The effect of chamber pressure on atomization and mixing of the propellants prior to combustion is probably minor compared with the effect on the combustion or reaction of the propellants. The atomization and mixing of impinging jets is probably affected very little by chamber pressure. The droplet size is controlled to the largest extent by the impact characteristics of the streams. Droplet vaporization rate, however, is retarded by an increase in ambient pressure but, again, this effect may be insignificant compared with the effects of pressure on the combustion process.

III, A, General (cont.)

In general, then, the effects of chamber pressure on the atomization, mixing, and vaporizations of propellants for impinging type injectors, should be minor compared with the effects on the combustion reaction.

B. CONTRACT AF 33(616)-5220

Considering the need for external application of forces to the liquid streams and for an efficient means of propellant mixing, basic splashplate and triplet elements were selected for evaluation. A degree of atomization and propellant distribution was to be achieved with a splashplate by means of direct impingement upon a metal surface. The resultant fan was then to be further atomized and mixed by means of impingement with a similar fan of the opposite propellant. In the case of the triplet element, complete atomization and propellant mixing was to be achieved by the impingement of two streams of one propellant on either side of a third stream of the other propellant (see Figure 10).

Available data from previous Aerojet work and from other sources were used where applicable. For example, published material on impinging fluid streams (Reference 1), was helpful in selecting, testing, and evaluating the large-thrust elements. Nonreactive tests were conducted with each of the two basic spray-elements, in which parameters such as splashplate contour, orifice size, entrance shape, pressure drop, and impingement angle were varied. Selection of the configurations for the reactive tests was based on analysis of the spatial distribution, mixture-ratio data, and known information on the propellant properties and mode of combustion.

III, B, Contract AF 33(616)-5520 (cont.)

The following paragraphs discuss the test installation, hardware, test procedures, and test results for both the nonreactive test program and the reactive (combustion) test program.

1. Phase I - Nonreactive Testing

a. Test Installation

Figure 11 illustrates the test fixture for holding the high thrust elements. The test liquids were pressure-fed from two 50-gal tanks to the element through the lines visible at the top of the figure. The relative efficiency of an injection-element was determined by the uniformity of the sprays mass distribution and mixture-ratio distribution. Measurements were made by using spray-sampling probes. The test liquids consisted of an aqueous solution of 0.5% by weight nitric acid to simulate fuel, and a 0.5% sodium hydroxide solution to simulate the oxidizer. The mixture-ratio of each sample was obtained by measurement of its pH value. The quantity of samples collected in the graduated cylinders in a known time interval indicated the weight flow rate. The samples were collected at a distance of 3 in. from the impingement point of each element; this location was selected because it was within the principal zone in which combustion occurred.

Two problems were encountered in previous investigations of liquid sprays where samples were taken in open-end tubes and collected in containers. The problems were (1) the effect of start-up and shut-down transients of the injector being tested on the sample volume and composition, and (2) the hydrodynamic effects at the entrance of the sampling tubes.

III, B, Contract AF 33(616)-5520 (cont.)

Because of the variation in mixture-ratio during starting and shut-down transients in the spray, a relatively long duration sample was required to minimize the effects of transients and obtain an accurate steady-state mixture ratio.

Disruption of the flow pattern upstream of the tube entrance causes an error in the flow rate in the tube. This is a hydrodynamic effect caused by a buildup of higher pressure inside the tube (because of partial flow stagnation) than outside the tube. This pressure difference results in the diversion of some of the spray around the tube entrance. Misalignment of the sample collecting tube from the direction of the liquid motion can also result in erroneous flow rates. To overcome these difficulties the sampling probe shown in Figure 12, Views D and E, was designed. This probe is a solenoid-actuated sample-tube in which the sample collection time is controlled by a ball valve located in the probe. Each probe is connected to a vacuum line so that the internal probe pressure can be controlled to minimize sample losses around the tip of the probe.

High-speed microflash photographs were taken to indicate the size, shape, wave frequency, and direction of travel of the spray. The photographs also aided in selecting the locations of the sampling probes to eliminate misalignment.

b. Spray Sampling Procedures

Preliminary experiments were conducted to determine the effectiveness of the vacuum system in minimizing hydrodynamic effects at the probe tip. In these experiments, a stream of water was directed downward directly to the inlet of each of three types of probes; the flow rate was constant in each case.

III, B, Contract AF 33(616)-5520 (cont.)

Figure 12, View A, shows a 1/4-in. tube (without the ball-valve feature, but with a tapered wall at the entrance), operating with the sample-collection cylinder open to atmospheric pressure; it collected 75% of the stream of water. Figure 12, View B, shows the specially-designed sample probe operating with the sample-collection cylinder open to atmospheric pressure; 33% of the stream was collected. Figure 12, View C, shows the sample probe operating under vacuum collection (approximately 2 psia pressure); the entire stream of water was collected.

It was believed that the straight tube with vacuum flow conditions would be as effective as the sample probe; however, this sampling method would not have the ball-valve feature which permits accurate control of the sampling duration. Thus, the vacuum-aspirated ball-valve-controlled sampling system was selected for use in this program.

Samples taken during the nonreactive spray tests were collected over an average duration of 1.2 sec. This short interval was necessitated by the large liquid-flow rates and the comparatively small capacity of the two propellant supply tanks (50-gal each). The injector element flowed approximately 40 sec; it had a 10-sec startup transient and a 5-sec shutdown transient, with steady-state flow for 25 sec. The samples were taken during the steady-state flow period. The following parameters were measured for each sample; the collection time duration, sample volume, liquid temperature, and pH (hydrogen ion concentration) of the sample (see Figure 13). From these parameters, the flow rate and mixture ratios of the spray at the sample points were determined.

III, B, Contract AF 33(616)-5520 (cont.)

Each of the elements tested produced a fan-shaped propellant distribution of elliptical cross-section. It was determined that the bulk of the spray could be sampled within a boundary determined by this one fan shape. Figure 14 illustrates the coordinate system used for locating and identifying the sampling positions, which lie on the surface of a semisphere having a 3-in. radius. The α angles defined the angular rotation of the sample points about the minor or α axis of the spray, and were a measure of the arc length along the major or β axis. The β angles defined the angular rotation of the sample points about the β axis, and were a measure of the arc length along the minor or α axis of the spray. Samples were taken at intersections of the great circles for $\alpha = 0°, \pm 10°, \pm 20°, \pm 35°, \pm 50°, \pm 90°$, and $\beta = 0°, \pm 7\text{-}1/2°$, and $\pm 15°$. Thus, comparative surveys of the various injector-element sprays were based on 55 spray-sample points.

The resulting mixture ratio and flow rate were determined at each of the sample points. From these data, the mixture ratio distribution and flow rate distribution were determined. From a comparison of the collected data, a reactive element configuration was recommended.

c. Test Results

Four injector designs of the splashplate type were evaluated and over 200 spray samples were taken. Several other designs were given preliminary design consideration. A triplet design was selected for use in the reactive test phase of the program, as it produced a more uniform mixture ratio distribution throughout the spray fan.

III, B, Contract AF 33(616)-5520 (cont.)

The flow rates and pressure drop for the various elements tested during Phase I are listed below.

Injector	$\dot{w}_f$ (lb/sec)	ΔP_f (psi)	$\dot{w}_o$ (lb/sec)	ΔP_o (psi)	M.R.	Type Injector
AIT Mod 1	9.7	91	9.7	90	1.0	Splashplate
AIT Mod 2	9.7	68	9.7	26	1.0	45° Triplet
AIT Mod 2A	9.7	68	9.7	26	1.0	60° Triplet
AIT Mod 2B	9.7	68	9.7	26	1.0	35° Triplet

The basic splashplate design, shown schematically in Figure 15, consisted of a jet from a single orifice impinging on a contoured plate. The distribution of liquid, and to some degree the droplet size in the resulting spray fan, was determined by the shape of the plate. Splashplate element injecting fuel and oxidizer can be operated adjacent to each other, or spaced apart, as desired.

Figure 16 illustrates the splashplate element tested. This element incorporates the use of two contoured metal plates placed back-to-back so that fuel and oxidizer impinge on either plate. The resultant fans are then impinged along the circumference of the adjacent plates. As shown in Figure 16, the oxidizer and fuel orifices have a comparatively high length/diameter ratio (L/D). Each of the orifices has a contoured inlet followed by a short threaded section. The contoured inlet reduces the tube entrance pressure losses and the threaded section decreases the tube length required to produce a stream with more fully developed turbulent flow (References 1 and 2). Thus, any discontinuities in the spray form are not attributable to the injection process, and problems introduced by manifolding and feed-line configurations are avoided.

III, B, Contract AF 33(616)-5520 (cont.)

Figure 17 is a photograph of the fan from the splashplate element, AIT Mod 1. View A is taken perpendicular to the α axis; View B is perpendicular to the β axis. The frequency of the radial fluctuations visible in View A is approximately 250 cps. Figure 18 shows the flow rate and mixture ratio distribution of this element. (The relatively large amount of scatter in the data at $\beta = 0°$ is due to fluctuations of the liquid spray fan.) The average deviation of the data from the $\beta = 0°$ line is $\pm$ 16.3%. The mixture ratio for this element varies from 0.5 to 1.2; comparison with data from triplet-element data showed this to be relatively poor distribution. Therefore, the element was not selected for reactive testing.

Two other contoured splashplate-type injector element designs were considered but not tested because of the poor performance of the AIT Mod 1 element.

Figures 19 and 20 illustrate the basic triplet element designated as the AIT Mod 2 type. Contoured inlet and threaded orifice inlet sections were used. Three triplet elements were tested, each with a different included angle (ϕ) between the outer and the central orifices (Figure 19) to evaluate the effects of impingement-angle variations.

Figure 21 shows the spray from the triplet element, AIT Mod 2. View A is perpendicular to the α axis and View B is perpendicular to the β axis. The degree of atomization for the triplet is noticeably better than it is for the splashplate (Figure 17).

III, B, Contract AF 33(616)-5520 (cont.)

Figure 22 shows the flow rate and mixture ratio distribution of the AIT Mod 2 element. The average deviation of the data points from the flow rate line of $\beta = 0°$ was $\pm$ 5.35%. Neglecting the one extraneous point at $\alpha = 0°$, the average deviation was $\pm$ 3.4%, which was within the accuracy of the sample collecting system. Inspection of Figure 22 revealed a range of this distribution was attributed to a minor misalignment (0.008 in.) in the orifice center lines. This was considered to be a relatively good mixture-ratio distribution and was comparable to that of the XLR87-AJ-1 thrust-chamber injector.

Figure 23 illustrates the flow rate and mixture ratio distribution of the AIT Mods 2A and 2B injection-elements. It can be observed that the AIT Mod 2 injector had a more uniform mixture-ratio distribution than the other two injectors. For this reason, it should result in a higher performance level.

The data indicated that, at an impingement half angle (ϕ) of 45°, the axial fluctuation of the spray-fan was at a minimum. Variation in ϕ had little effect on the width and thickness of the fan.

Figure 21 shows that the major portion of the spray along the β axis is between the angles $\alpha = +45°$ and $\alpha = -45°$. Only a slight amount of spray was located beyond the angles $\alpha = 90°$ and $\alpha = -90°$, minimizing tendencies toward injector-face erosion. This effect would be further minimized by the proper design of an injector-face cooling manifold.

III, B, Contract AF 33(616)-5520 (cont.)

Based on the uniformity of mixture-ratio distribution, flow-rate distribution, and fan shape, the triple element (AIT Mod 2), with an impingement half angle of $\phi = 45°$, appeared to be the best of the elements tested. It was therefore selected for the Phase II combustion tests.

2. Phase II - Reactive Testing

a. Test Installation

A schematic diagram of the test installation which was used for reactive tests with N_2O_4/UDMH and N_2O_4/Aerozine-50 is shown in Figure 24.

The injectors were tested in the transparent chamber shown in Figure 25. The chamber design parameters are:

Characteristic Length (L*)	- 65 in.
Contraction Ratio (A_c/A_t)	- 2.67
Throat Area (A_t)	- 18.0 in.^{2}
Expansion Ratio (A_e/A_t)	- 3.0

For maximum flexibility, the chamber was designed so that the injector and windows may be changed by removing support braces.

High-speed color motion pictures of the combustion tests were taken through a 2-in.-thick Plexiglas window. In order to minimize the costs of changing windows between tests, a 1/4 in.-thick Plexiglas liner for the inside of the window was provided. This liner could be inexpensively replaced after each test. A steel liner was used on the side of the chamber opposite the Plexiglas window. The steel liner was replaced with Plexiglas on two tests to permit

II, B, Contract AF 33(616)-5520 (cont.)

photograph of the gases in the throat region using a back-lighting technique. The window liner can also be made of quartz or Vycor should corrosive propellants be tested in this chamber. The window braces are movable, and were placed in positions which would avoid blocking of the view of particularly interesting portions of the combustion zone such as the injector face and throat.

The thrust chamber was mounted horizontally on a test stand and the thrust was measured with a pressure-loaded thrust jack.

Separate propellant valves were used to control the flow of fuel and oxidizer. Each valve was nitrogen-actuated and equipped with automatic prefiring and post-firing nitrogen purges. The actuation times were sequenced to provide about a 100-millisec oxidizer lead. Relative actuation times for the two valves were determined from hydraulic-flow tests in which the valves were actuated and the initiation of flow from the injector was photographed using a high-speed picture camera.

The propellant flow rates were measured with a turbine-type flow meter and a cavitating venturi. The cavitating venturi also served to prevent any pressure fluctuations which might occur from propagating from the thrust chamber to the propellant tanks.

b. Injectors for Reactive Tests

The designs of the injector elements used for the reactive tests were based on data from the nonreactive tests conducted under Phase I of this program, and on criteria developed by JPL (Reference 3). The results of Phase I testing indicated that the triplet type of injector was most suitable for a high-thrust injector element because of better propellant dispersion and mixing.

III, B, Contract AF 33(616)-5520 (cont.)

The triplet elements used for these tests consisted of two fluid jets of oxidizer impinging on a jet of the fuel at a common point. The axes of the jets are located in the same plane, and the angle between jets can be adjusted to achieve a limited control of the resulting spray fan.

The physical characteristics of the elements designed for reactive testing under Phase II of the program are summarized in Table 2.

This uncooled triplet type of injector was fabricated by inserting three standard tubes or pipes into a drilled base plate and brazing them into place. The tubes and base plate were fabricated from Type 347 stainless steel. To reduce injector face erosion caused by propellant splash-back, the thickness of metal surrounding the orifices was minimized. The portion of the tubes which project beyond the face were cooled by the propellants flowing through the tubes. Injectors that would incorporate several of these elements could be easily manifolded to provide regenerative cooling of the injector face.

c. Test Results

Two satisfactory tests were conducted using N_2O_4/UDMH and eleven satisfactory tests were conducted using N_2O_4/Aerozine-50. The results of these tests are presented in Table 1. Steady-state experimental c * values up to 97.8% of theoretical (based on shifting equilibrium values) were achieved at optimum mixture ratio and 500 psia chamber pressure using a single triplet injector element operating at a sea level thrust of 13,000 lb with N_2O_4/Aerozine-50.

III, B, Contract AF 33(616)-5520 (cont.)

(1) N_2O_4/ UDMH

Two satisfactory tests were conducted with the AIT Mod 2C injector (Figures 26, 27). The results are presented in Table 1. Figures 4, 5, and 6 show c*, I_{sp}, and C_f, respectively, as functions of mixture ratio. The experimental data were adjusted to a 500-psia plenum chamber pressure as outlined in paragraph III, B, 2, c, (4), (a).

As indicated in Figure 4 which shows c* as a function of mixture ratio, the maximum efficiency obtained is 93.5% of the theoretical c*. The maximum experimental c* occurs at a mixture ratio well above the theoretical optimum of 2.15. Because of the greater interest in the N_2O_4/Aerozine-50 propellants, no attempt was made to optimize the injector for N_2O_4/ UDMH to obtain high performance at the theoretical optimum mixture ratio.

Theoretical and experimental data for I_{sp} vs mixture ratio are presented in Figure 5. The maximum efficiency is 89% of the theoretical value at a mixture ratio of 3.06, which again is well above the theoretical optimum mixture ratio of 2.0.

Theoretical and experimental values of C_f vs mixture ratio are presented in Figure 6. The measured values are about 75% of the theoretical values.

III, B, Contract AF 33(616)-5520 (cont.)

(2) N_2O_4/Aerozine-50

Eleven satisfactory tests were conducted with N_2O_4/Aerozine-50. The results are presented in Table 1. Figure 1, 2, and 3 show c*, I_{sp}, and C_f, respectively, as a function of mixture ratio. Steady-state experimental c* values up to 97.8%, and I_{sp} values up to 92.2% of the theoretical shifting equilibrium data, were obtained. The experimental data have been adjusted and compared to a 500 psia plenum chamber pressure.

The initial tests (D-217-LJ-3 through -6) were conducted with a triplet injector having a 90° included angle, L/D of over 15, and a smooth, round orifice (pattern AIT Mod 2C-1, see Figure 28). The orifice sizes were calculated using Equation (1), and the flow rates based on 500-psia chamber pressure and a mixture ratio to produce maximum c*.

The following equation (Reference 3) relates injector design parameters and the physical properties of the propellants to the uniformity of mixing for a two-on-one impinging injector (triplet).

$$\left(\frac{\dot{w}_o}{\dot{w}_f}\right)^2\left(\frac{\rho_f}{\rho_o}\right)\left(\frac{A_f}{A_o}\right)^{1.75} = C \qquad (1)$$

where:

A_o = total orifice area of both oxidizer orifices
A_f = fuel orifice area
C = empirical constant

III, B, Contract AF 33(616)-5520 (cont.)

The value of C in the above equation is dependent on the total impingement angle. For triplets, the C value varies from 0.85 with a 60° included angle to 0.70 with a 90° angle.

Knowing the desired fuel flow rate, density, and pressure drop, a fuel-orifice size is calculated. The equation is then solved for the proper oxidizer-orifice area.

Combustion efficiencies of about 78% of the theoretical were obtained using the AIT Mod 2C-1 injector. It was noted that a change in the mixture ratio occurred during the tests as a result of the opening characteristics of the propellant valve. The number beside each data point in Figures 1, 2, and 3, and Table 1, indicates the test number. The letters "A" and "B" represent data at two different mixture-ratios, each at a different steady-state portion of a test. The "A" point shown in the table and figures was obtained from data taken near the beginning of the test when the oxidizer is flowing at full flow rate and the fuel flow is partially throttled. This results in a slightly higher mixture ratio than is obtained where data point "B" is taken. This technique provided maximum information from a minimum number of tests.

Analysis of this data indicated that a higher percentage of theoretical c* was obtained at the higher mixture ratios. Subsequent tests were then conducted at increasing mixture ratios. Performance improved: about 94% of theoretical c* was achieved at a mixture ratio of 2.4 (Figure 1). A ratio of the oxidizer-momentum to fuel-momentum was plotted as a function of the percent of theoretical c* obtained, and it was noted that performance was directly proportional to the momentum ratio over the mixture ratio range investigated (Figure 29). An injector was designed and tested in which the orifice sizes were

III, B, Contract AF 33(616)-5520 (cont.)

calculated to provide the proper momentum value at the optimum mixture ratio (AIT Mod 2C-1A, Figure 30). The performance for this injector at the optimum mixture ratio was over 95% of theoretical c* (Test D-217-LJ-7). A low frequency fluctuation in chamber pressure (± 3% or more) was present in all tests. An additional test was conducted (-8) in which the momentum ratio was held constant and the L/D ratio was decreased to 4.5 (oxidizer) and 8.0 (fuel) to investigate the effect of L/D on chamber pressure oscillations. (Smaller L/D orifices are also more readily adapted to full-scale injector design). No change in the performance or the chamber oscillations were measured.

The included impingement angle was then decreased from 90° (AIT Mod 2C-8) to 75° (AIT Mod 2D-8) and then to 60° (AIT Mod 2E-8) without a significant change in performance, although an increase in the amplitude of the chamber pressure oscillations was noted.

A test was made with the 90°-impingement-angle triplet-injector (AIT Mod 2C-8A) in which the orifices were threaded internally. The purpose of this test was to determine if more fully developed turbulent flow would have an effect on the oscillations or performance; no change was noted.

In order to increase the surface contact area between the oxidizer and fuel streams, an injector having elliptical oxidizer orifices and a rectangular fuel orifice was designed (AIT Mod 2F-8, Figure 10).

III, B, Contract AF 33(616)-5520 (cont.)

The advantage of this configuration can be seen from Figure 31. In order to relate the increase of momentum of the oxidizer streams to that of the fuel streams, it was necessary to decrease the ratio D_o/D_f as shown in View A. This increased the portion of the fuel stream (shaded area) which did not directly impinge on the oxidizer streams. As a result, this portion of the fuel stream was forced outward without being adequately mixed with the oxidizer. This condition can result in a degradation of performance.

In the design depicted in View B, Figure 31, the width of the oxidizer streams (h_o) was made equal to the width of the fuel stream (h_f). The surface contact area was increased, and the unmixed area in the pattern was eliminated.

The results of the two tests with this injector (AIT Mod 2F-8) (D-217-LJ-12 and -13) are shown in Table 1. A slightly higher performance (95% of theoretical c*) was achieved. The chamber pressure fluctuations were slightly less than they were for previous tests.

Pertinent dimensions of the injectors which have been discussed above are listed in Table 2.

Figure 2 shows experimental and theoretical I_{sp} as a function of mixture ratio for N_2O_4/Aerozine-50. This curve indicated that tests D-217-LJ-12 and -13 have the highest performance. The I_{sp} values are believed to be somewhat degraded by the rectangular cross-section of the nozzle throat and expansion-section. The boundary-layer or edge-effects of this nozzle would tend to lower C_f values, which in turn would result in lower I_{sp} values than would occur with a conventional nozzle.

III, B, Contract AF 33(616)-5520 (cont.)

(3) Accuracy of Data

(a) Theoretical Data

The theoretical data (Table 3) was obtained from IBM 704 computer programing based on (1) shifting equilibrium, (2) nozzle-expansion area ratio of 3.0, (3) λ factor of 0.977 (correction for nozzle half-angle of 17.5°), and (4) plenum chamber pressure of 500 psia.

(b) Experimental Data

The experimental data for c*, I_{sp}, and C_f were calculated from the following standard rocket-engine-performance equations:

$$c^* = \frac{P_c A_{tg}}{\dot{w}_t}$$

$$I_{sp} = \frac{F}{\dot{w}_t}$$

$$C_f = \frac{F}{P_c A_t}$$

Although a comprehensive statistical analysis of the errors was not attempted, the over-all accuracy of the above parameters for a 1-sigma deviation is estimated to be $\pm$ 2%.

III, B, Contract AF 33(616)-5520 (cont.)

1 Thrust

The maximum deviation in thrust of the calibrated thrust stand is estimated to be ± 0.5%.

2 Flow

The flow rates shown in Table 1 were based on measurements by Potter flow meters. The accuracy of flow measurement is estimated to be within ± 1%. Venturi meters were also used, and ± 3/4% correlation to Potter data was achieved.

3 Throat Area

The rectangular nozzle-throat area was determined by multiplying the average of four vertical measurements by the average of two horizontal measurements. Erosion of the Plexiglas wall results in an increase in the throat area during tests. The throat area at the time at which the data is reduced is obtained by a linear interpolation of the initial throat area and the final throat area.

4 Chamber Pressure

Wiancko pressure transducers are used to obtain the chamber pressure; their accuracy is estimated to be within ±1-1/2%.

III, B, Contract AF 33(616)-5520 (cont.)

(4) Sample Calculations

Test No. D-217-LJ-4a is used here to illustrate the procedure employed in calculating the experimental performance. The test data is as follows:

Data-point time	= 0.67 sec after firing-switch actuation
$P_{c\text{-throat}}$	= P_{c-2} = 384 psia
A_t	= 18.15 $in.^2$
g	= 32.2 ft/sec
$\dot{w}_t$	= 56.8 lb/sec (from Potter flow meters)
M.R.	= 1.56
Thrust, F	= 9,870 lb
Chamber Contraction Ratio (A_c/A_t)	= 0.375

(a) Correction of Chamber Pressure

Correction of $P_{c\text{-throat}}$ to plenum chamber pressure is made using data contained in Reference (4), which presents curves for $P_{c\text{-plenum}}/P_{c\text{-throat}}$ as a function of contraction ratio and gas specific heat, γ. The value of γ for these propellants is approximately 1.20. Therefore,

$$P_{c\text{-plenum}} = \frac{P_{c\text{-throat}}}{0.97} = \frac{384}{0.97} = 396 \text{ psia}$$

III, B, Contract AF 33(616)-5520 (cont.)

(b) Characteristic Exhaust Velocity

At a P_c of 396 psia,

$$c^* = \frac{P_c A_t g}{\dot{w}_t} = \frac{396 \times 18.15 \times 32.2}{5.68} = 4,084 \text{ ft/sec}$$

For correlation with the theoretical c*, the experimental c* is adjusted to a P_c of 500 psia using Figure 32 which shows the c* correction as a function of P_c. Thus the experimental c*, adjusted to c* at 500 psia, is given by

$$c^*_{adj} = 4,084 + 13 = 4,097 \text{ ft/sec}$$

The theoretical c*, obtained from Figure 1 is 5,760 ft/sec. The experimental c* is thus 4,097/5,760 (100) = 71.1% of the theoretical value.

(c) Specific Impulse

At P_c = 396 psia,

$$I_{sp} = \frac{F}{\dot{w}} = \frac{9,870}{56.8} = 174 \text{ sec}$$

The experimental I_{sp} is adjusted to a 500-psia P_c using Figure 33. Therefore,

$$I_{sp, adj} = 174 + 5.0 = 179 \text{ sec}$$

III, B, Contract AF 33(616)-5520 (cont.)

The theoretical I_{sp} of 254 sec, obtained from Figure 2 includes the λ factor (correction for 17.4° half-angle) of 0.977. Therefore, the experimental I_{sp} at a 500-psia P_c is 179/254 (100) = 70.5% of the theoretical value.

(d) Thrust Coefficient

At a P_c of 396 psia,

$$C_f = \frac{F}{P_c A_t} = \frac{9,870}{396 \times 18.15} = 1.373$$

The experimental C_f at P_c = 396 psia is adjusted to P_c = 500 psia using Figure 34. This results in a $C_{f,\,adj}$ at 500 psia of 1.373 + 0.025 = 1.398.

The theoretical C_f of 1.420, obtained from data shown in Figure 3 includes the λ factor of 0.977. The experimental C_f is thus 1.398/1.420 (100) - 98.6% of the theoretical value.

(e) Presentation of Data

The experimental and theoretical curves (Figures 1, 2, and 3) include the corrections and adjustments discussed above. In all calculations, it was assumed that (1) there were no heat-transfer losses to the thrust-chamber walls, and (2) there was isotropic expansion through the nozzle.

III, Technical Discussion (cont.)

C. CONTRACT AF 04(647)-521

1. Phase I--Performance Configuration

Table 4 presents the performance data for the 10 tests in Phase I and Figures 35 and 36 show c* and I_{sp}, respectively, versus mixture ratio. c* values up to 5,930 ft/sec and I_{sp} values up to 94.5% of theoretical were obtained. The first nine tests were conducted for a 1.5 sec duration; the last test was conducted for a 3.6 sec duration. Smooth hypergolic ignition and stable steady-state operation (P_c oscillations less than ±3.5%) was achieved.

Two tests were conducted using the smallest thrust chamber, L* = 51 in. A minor gas leak was noticed in the flange seal between the injector and chamber. This was corrected by using a copper gasket in place of the Durabla gasket. Low frequency pulsing of a magnitude approximately ±10% of P_c was noticed.

Two additional tests were conducted using a thrust chamber having an L* = 59 in. The combustion performance was low, approximately 90% of theoretical.

Six tests were conducted with the large thrust chamber, L* = 100 in. and L' = 25 in. Excellent combustion performance was achieved, c* values up to 103.8% of theoretical with chamber pressure oscillations less than ±3.5% of P_c. Three of these tests resulted in c* values over 100% of theoretical.

This group of tests has shown the importance of chamber geometry on the combustion performance when large-thrust-per-element injectors

III, C, Contract AF 04(647)-521 (cont.)

are used. Although the highest performance tests were obtained with a thrust chamber of L* = 100 in. and L' = 25 in., it is believed that similar results would be obtained with a properly designed smaller thrust chamber, L* = 60 in. This could be done by decreasing the inside diameter of the existing chamber from 6-7/8 in. to 5-1/2 in. and decreasing L' length from 25 in. to 23 in.

No hardware difficulties were encountered, except the minor seal leakage on the first two tests during this program. The last test was conducted for a 3.6 sec duration. No injector or chamber erosion was noted although the throat of the copper nozzle was slightly eroded.

2. Phase II--Combustion Stability Characteristics

Test data from the combustion stability investigation, is presented in Table 5, and the composition of fuel used for each test is shown in Table 6. The first 13 tests in the series were conducted with the injector spray fan parallel to the chamber wall. Tests D-223-LJ-1 to -7 were made at the "design point," mixture ratio 2.0, and same flow rates ($\dot{w}_t$ approximately 45 lb/sec), but with varying chamber pressure (accomplished by changing nozzle throat size). Tests at lower P_c were run to permit the use of a Plexiglas window. This allowed photographic observation of the instability phenomena and made possible a more thorough investigation of the combustion instability. Both single- and multi-front instability were observed. Marginal to poor stability was exhibited under these test conditions.

Tests 8 to 11 were conducted at a mixture ratio of approximately 1.60 and total flow rates of 33 and 50 lb/sec. P_c was maintained constant through this series. All four of these tests were spontaneously unstable before any pulses were fired. Tests 8 and 9, were terminated prematurely by the

III, C, Contract AF 04(647)-521 (cont.)

malfunction time due to slow filling of the injector manifolds at the low (33 lb/sec) flow rate. The timer was reset for a longer duration for Tests 10 and 11, however, the instability characteristics remained the same. Tests 12 and 13 were run at higher total flow conditions (52 and 55 lb/sec) and at mixture ratios of 2.4 and 2.0, respectively. Stability performance at these conditions was similar to that obtained at the design mixture ratio point.

The stability contour for the parallel spray fan orientation is found in Figure 9 (B). Straight lines on the figure indicate lines of constant mixture ratio. Each point is labeled according to the minimum pulse which induced tangential instability. Envelopes are drawn around those points having similar syability characteristics and a stability index is indicated for each region.

Tests 14 to 23 were conducted with the spray fan perpendicular to the chamber wall. Results of these tests indicate moderate to good stability. Tests 14 and 15 (2.00 and 2.40 mixture ratios and total flows of 55 and 53 lb/sec, respectively) were stable on a pulse charge of V-20.[1] The conditions of these tests were repeated on Tests 20 and 21 to ascertain the effect of pulse charges up to V-80. At the 2.0 mixture ratio and 55 lb/sec flow, combustion instability was sustained with a minimum induced charge of V-80 (Test 20). Test 21 was stable throughout the entire range of available pulse charges. (2.4 M.R., $\dot{w}_t$ = 53 lb/sec). The stability contour for the perpendicular-to-wall spray fan orientation is presented in Figure 9 (D). Following the last test (23), injector tube erosion was noted on those injection tubes which faced the tangential instability wave. The severity of the erosion indicated that it most likely occurred on the last several tests rather than the last test, even though previous erosion was not specifically noted by test personnel. A typical eroded element is shown in Figure 37.

[1] The -20 refers to a smokeless powder charge weight of 20 grains

III, C, Contract AF 04(647)-521 (cont.)

Sufficient test data had been accrued so no attempt was made to repair the elements and continue the program.

3. Phase III--Comparison of Multiple Element With Single Element Performance

Table 7 presents the performance data for the ten tests conducted with the uncooled thrust chamber assembly which is shown in Figure 11. c* values from 95.9 to 100% of theoretical and I_{sp} values from 90 to 97% of theoretical were obtained. Smooth hypergolic ignition and stable steady-state operation (P_c oscillations less than $\pm$4.5%) was achieved. The nominal test duration was 1.3 sec.

Tests D-225-LJ-1 through -4 were conducted with the 59-in. L* chamber and the injector elements oriented so that the spray fan was perpendicular to the chamber wall. The spray pattern and element orientation is shown in Figure 38. The c* values varied from 5,510 to 5,722 ft/sec. At the end of Test 4, the nozzle was removed and three "hot spot" areas were noted in the thrust chamber. These areas were located equidistant between the element spray fans and extended the length of the chamber. This indicates that the chamber wall areas that evidence the direct propellant spray fan remain cooler than the areas which are between these spray fans.

Tests 5 and 6 were conducted with the 50-in. L* chamber and with the injector elements oriented so that the spray fan was tangential to the chamber wall as shown in Figure 38. c* values of 5,606 and 5,589 ft/sec were achieved which is slightly less than the values achieved with a perpendicular spray fan orientation.

III, C, Contract AF 04(647)-521 (cont.)

Tests 7 and 10 were conducted with a 59-in. L* and 51-in. L* chamber, respectively, and with the injector elements oriented so that the spray fan was at a 45° angle to the chamber wall as shown in Figure 38. This element orientation provides slightly higher (approximately 1% of c*) performance than the other two element orientations tested. It is speculated that the reason for this lies in the fact that the spray fan impinges on a larger wall area.

One test (8) was conducted with the 100-in. L* chamber. No increase in performance was noted over the smaller L* chamber tests. The results of this program have shown that a three-element injector achieves complete combustion in a 51-in. L* chamber as shown in tests 9 and 10. Although no tests were conducted with L* chambers less than 51 inches, it is reasoned that good combustion can be achieved with lower L* chamber, possibly as low as 40 inches.

The interaction effects between adjacent elements is seen by comparing the single-element performance in Phase I and the multiple-element performance in Phase III in the same size thrust chamber. This is shown in Figure 39 with the 51-in. L* chamber and in Figure 40 with the 59-in. L* chamber. The increase in performance with three elements is approximately 7.5% in I_{sp} for the 51-in. L* chamber and approximately 6.3% of I_{sp} for the 50-in. L* chamber. The reasons for this higher performance are that more efficient use of the chamber cross-sectional area near the injector face is realized with three elements and also the interaction of adjacent spray fans promotes better combustion in a smaller volume.

No hardware difficulties were encountered during this program. Slight copper nozzle erosion occurred in tests of greater than 1.3-sec duration.

III, C, Contract AF 04(647)-521 (cont.)

Excellent sealing and no propellant corrosion was obtained with the Teflon O-rings used on the element pods. An alternate O-ring material (Fluorosilicone Elastomer) was used in Test 10 on one element pod with good results. The advantage of this elastomer over Teflon is that it is more pliable and can be handled similar to neoprene O-rings. A Kel-F elastomer O-ring was used for the chamber-injector interface seal with good results although the heat soak-back after firing slightly burned the material so that it required changing after every three tests.

D. VARIABLE THRUST APPLICATION

Current work at Aerojet has resulted in a simple throttling injector design that incorporates a single triplet injection element in which throttling of the propellant flow is accomplished by a single throttling pintle located in each injection orifice. The design involves the established hydraulic-needle-nozzle principle, Reference 5. The success of this design results from the ability to obtain high combustion performance using a large-scale, single-element injector.

1. Injector Design

Two injector designs which represent promising characteristics are shown in Figures 41 and 42. The main difference between the two designs lies in the actuation system; one incorporates a linkage chain (push-pull), the other a gear train. Actual flow control in both injectors is accomplished by a set of pintles, utilizing the hydraulic-needle-nozzle principle. This principle, described in detail in Reference 5, is used in the design of hydroelectric-power generators to control the uniformity and size of the water jet impinging on the Pelton Water-Wheel over a wide variation of flow rates.

III, D, Variable Thrust Application (cont.)

This nozzle design efficiently converts constant potential heat (tank pressure) to kinetic head (velocity) which is required for good combustion performance. The injector flow velocities (and areas) were determined empirically from data gathered on the Large-Thrust-Per-Element program under Contract AF 33(616)-5220. Water spray characteristics of a 1,000- to 10,000-lb injector design is shown in Figure 43.

The actuation of the injector mechanism is provided by an electrical actuator, which is composed of an electrical high-speed motor, gear reduction, clutch, brake, and rotary potentiometer. An electrical actuation system is used over a hydraulic system because of design simplicity in space environment.

a. Push-Pull Cable Concept

The push-pull cables are capable of taking compression loads, and are used to drive the oxidizer pintles only. The three pintles are secured to each other by a formed channel section and operate as a single unit. Movement of the injector pintles is accomplished by the electrical actuator cam which is engaged with a pin in the female end sleeve of the fuel pintle.

(1) Pintle Operation

An oxidizer lead is obtained by a built-in delay in the fuel pintle which is composed of the plug end, female end sleeve, compression spring, and actuating pin. In the injector fully-closed position, the spring is compressed by the female end sleeve.

Operation of the injector from the closed position to an operating position is accomplished by energizing the electrical actuator. Under

III, D, Variable Thrust Application (cont.)

the impetus of the cam, the female end sleeve starts its lineal movement, and in so doing raises the two oxidizer pintle plugs from their respective seats, thereby initiating flow. The plug end of the fuel pintle remains in the closed position, held there by the force of the compression spring.

After oxidizer flow has been accomplished the pintle actuating rod in the female end sleeve engages the fuel pintle, lifts it from its seat, thereby establishing fuel flow.

The injector can now be adjusted and maintained, at the desired thrust level.

(2) Pintle Operation-Shutdown

The same sequence of operation is repeated, in reverse, when thrust chamber shutdown is desired.

(3) Injector Actuation Time

The time required to actuate the injector pintles from completely closed to fully open is determined by the lead of the cam groove. Degree of injector opening may be lineal, or can be varied by the proper cam groove design.

b. Gear Train Concept

In the gear train concept, five bevel bears are utilized to drive the pintles. The gears used consist of a basic drive gear - gear train and fuel pintle, two idler gears, and a driven gear for each oxidizer pintle. The general configuration of the gear train concept is similar to the push-pull cable,

III, D, Variable Thrust Application (cont.)

except that movement of the injector pintles is accomplished through threaded gears rotating on nonrotating threaded pintle shafts. The use of idler gears in the gear train resulted in smaller diameter gears inasmuch as the distance of the fuel/oxidizer pintles at the driven ends became quite excessive.

(1) Pintle Operation

The pintle operation is basically identical to the design used in the push-pull cable configuration.

(2) Injector Actuation Time

The time required to actuate the injector pintles, from completely closed to fully open, will be determined by the rotary speed of the output shaft of the electrical actuator.

2. Restart Capability

The main requirements for satisfactory ignition are: (a) ignition delays must be sufficiently short so that "hard" starts do not occur. and (b) the system must be capable of reproducible starting transients. For the case of the highly hypergolic propellants such as N_2O_4/Aerozine-50 and IRFNA/UDMH, the reactivity and ignition characteristics are known ignition delays and range from 5 to 10 milliseconds, therefore, no start or restart problems are foreseen. On the other hand, nonhypergolic propellant combinations (for example, liquid oxygen and liquid hydrogen) require an external ignition source, such as a spark plug.

REFERENCES

1. A Correlation Between the Dynamic Properties of a Pair of Impinging Streams and Uniformity of Mixture Ratio Distribution in the Resulting Spray, Jet Propulsion Laboratory, Progress Report 20-209, 28 March 1956.

2. The Liquid Phase Mixing of a Pair of Impinging Streams, Jet Propulsion Laboratory, Progress Report 20-195, 6 August 1953.

3. A Criterion for Optimum Mixture Ratio Distribution Using Several Types of Impinging Stream Injectors, Jet Propulsion Laboratory Memorandum 30-5, 25 February 1959.

4. An Investigation of the Influence of Rocket-Chamber Configuration on Performance, Jet Propulsion Laboratory, Progress Report 1-82, 4 June 1952

5. Lionel S. Marks, Mechanical Engineers Handbook, 5th Edition, McGraw-Hill, 1951, p. 1299.

(End Text)

TABLE 1

LARGE THRUST PER INJECTION ELEMENT PERFORMANCE

Test No. D-217-LJ-	Mixture Ratio	P_c (1) (PSIA)	Pressure Oscillation (±% of P_c)	$\dot{W}_T$ (LB/SEC)	F (2) (LB)	C^* (3) (FT/SEC)	% of Theor. C^*	I_{SP} (3) (SEC)	% of Theor. I_{SP}	C_F (3)	% of Theor. C_F	ΔP_{OV} (4) (PSI)	ΔP_{FJ} (5) (PSI)	A_T (6) ($IN.^2$)	Data Point Time (SEC)	Test Duration (SEC)	Injector AIT Mod
TESTS WITH N_2O_4/UDMH																	
1A (7)	2.93	423	8.0	50.1	9970	4956	89.7	207	86.5	1.341	95.8	33	82	18.2	.72	1.69	2c
1B	2.11	384	20.0	54.0	9175	4226	74.3	183	73.2	1.382	97.8	32	143	18.	1.40	1.69	2c
2A	3.06	509	5.9	58.0	12,240	5141	93.5	211	88.6	1.324	94.6	45	97	1..2	.79	1.69	2c
2B	2.55	499	9.0	60.3	12,120	4876	86.5	202	82.1	1.333	95.3	44	135	18.3	.92	1.69	2c
TESTS WITH N_2O_4/AEROZINE 50																	
3A	2.09	480	4.2	55.4	11,100	5080	89.9	204	81.8	1.29[illegible]	.v		73	18.2	.63	1.18	2c-1
3B	1.94	465	8.0	56.0	11,160	4924	86.3	204	81.3	1.326	93.4	:	82	18.4	1.05	1.18	2c-1
4A	1.56	396	12.6	56.8	9870	4097	71.1	179	70.5	1.398	98.6	45	110	18.2	.67	1.17	2c-1
4B	1.58	391	30.7	56.0	9790	4149	72.1	180	70.8	1.398	98.5	43	106	18.4	1.12	1.17	2c-1
5A	2.22	500	7.0	61.8	12,640	4794	85.5	204	82.8	1.372	96.5	67	83	18.4	.62	1.18	2c-1
5B	2.16	489	8.2	62.0	12,600	4775	85.0	205	82.6	1.375	96.6	66	86	18.8	1.04	1.18	2c-1
6A	2.50	521	6.8	60.2	13,220	5145	93.7	217	89.3	1.362	95.8	68	67	18.4	.64	1.18	2c-1
6B	2.37	512	7.7	60.6	13,160	5149	92.9	216	88.0	1.354	95.5	67	73	18.8	1.06	1.18	2c-1

CONFIDENTIAL

TABLE 1 (cont.)

Test No. D-217-LJ-	Mixture Ratio	P_c (1) (PSIA)	Pressure Oscillation (±% of P_c)	$\dot{W}_T$ (LB/SEC)	F (2) (LB)	C^* (3) (FT/SEC)	% of Theor. C^*	I_{SP} (3) (SEC)	% of Theor. I_{SP}	C_F (3)	% of Theor. C_F	ΔP_{OJ} (4) (PSI)	ΔP_{FJ} (5) (PSI)	A_T (6) ($IN.^2$)	Data Point Time (SEC)	Test Duration (SEC)	Injector AIT Mod
TESTS WITH N_2O_4/AEROZINE 50																	
7A	2.26	521	7.5	56.6	11,880	5430	97.1	229	92.7	1361	95.6	70	26	18.3	.59	1.17	2C-1A
7B	1.71	515	5.5	56.0	12,650	5560	96.6	225	88.8	1307	92.3	69	25	18.8	1.02	1.17	2C-1A
8A	1.83	517	4.6	56.4	12,735	5440	94.7	225	89.1	1337	94.5	109	62	18.4	.65	1.17	2C-8
8B	1.77	494	7.5	56.0	12,535	5315	92.5	224	88.5	1360	96.1	106	65	18.7	.95	1.17	2C-8
9A	1.88	537	6.5	56.9	13,025	5610	97.9	227	90.2	1308	92.5	114	62	18.4	.70	1.17	2D-8
9B	1.79	506	15.2	56.5	12,145	5410	94.1	215	85.1	1279	90.4	108	65	18.8	1.04	1.17	2D-8
10A	1.85	498	5.4	57.8	12,655	5350	93.5	219	87.0	1320	93.3	116	66	19.2	.70	1.17	2E-8
10B	1.82	482	13.0	56.2	11,685	5450	95.0	209	82.8	1234	87.2	111	66	19.7	1.03	1.17	2E-8
11A	1.92	524	7.0	37.2	12,810	5435	95.0	222	88.4	1323	93.3	131	59	18.4	.66	1.17	2C-8A
11B	1.87	491	6.5	56.6	12,370	5275	92.1	219	87.0	1333	94.2	126	60	18.9	1.10	1.17	2C-8A
12A	1.94	527	5.5	56.50	13,160	5710	99.9	231	91.6	1310	92.4	166	69	19.0	.63	1.17	2F-8
12B	1.84	483	6.2	56.0	12,920	5425	94.6	232	92.2	1375	97.1	156	69	19.5	1.09	1.17	2F-8
13A	1.86	538	3.5	55.8	13,385	5715	99.6	237	94.1	1343	94.8	156	58	18.4	.65	1.17	2F-8
13B	1.79	512	3.9	55.3	12,950	5625	97.8	233	92.1	1338	94.4	150	60	18.9	1.09	1.17	2F-8

NOTES: (1) P_c IS MEASURED JUST PRIOR TO THE NOZZLE CONVERGING SECTION AND CORRECTED TO PLENUM CHAMBER PRESSURE BY DIVIDING BY .97.
(2) CHAMBER CONTRACTION RATIO, A_C/A_T = 2.67; NOZZLE EXPANSION RATIO, A_E/A_T = 3.0.
(3) C^*, I_{SP} AND C_F VALUES ARE CORRECTED TO 500 PSIA PLENUM CHAMBER PRESSURE TO FACILITATE COMPARISON WITH THEORETICAL.
(4) PRESSURE DROP ACROSS OXIDIZER ORIFICES . IT IS ESTIMATED THAT OVER 50% OF THIS DROP OCCURS ACROSS THE INLET FITTING OF THE TUBE.
(5) PRESSURE DROP ACROSS FUEL ORIFICE. IT IS ESTIMATED THAT OVER 50% OF THIS DROP OCCURS ACROSS THE INLET FITTING OF THE TUBE.
(6) A_T BASED ON INTERPOLATION BETWEEN INITIAL AND FINAL THROAT AREAS.
(7) LETTERS A AND B CORRESPOND TO DATA TAKEN AT INITIAL AND FINAL STEADY STATE OPERATION RESPECTIVELY;

CONFIDENTIAL

TABLE 2

LARGE-THRUST-PER-ELEMENT INJECTORS USED FOR PHASE II (REACTIVE) TESTS

AIT Mod -	Impingement Angle, deg	OXIDIZER I.D. (in.)	OXIDIZER L/D	FUEL I.D. (in.)	FUEL L/D	Used for Test No. D-217-LJ-	Spray Charac. Figure No.	Notes
2C	90	0.742	13.5	0.674	12.0	1,2	98	(2)
2C-1	90	0.710	14.0	0.742	11.0	3,4,5,6	99	(3)
2C-1A	90	0.652	15.0	0.825	10.0	7	101	(4)
2C-8	90	0.620	8.0	0.870	4.5	8	160	(5)
2D-8	75	0.620	8.0	0.902	4.4	9	161	(6)
2E-8	60	0.620	8.0	0.902	4.4	10	162	(7)
2C-8A	90	0.620	8.0	0.870	4.5	11	163	(8)
2F-8	90	0.620	8.0	0.870	4.5	12,13	81	(9)

NOTES:
(1) Tests D-217-LJ-1 and -2 were performed with N_2O_4/UDMH; tests -3 through -13 were performed with N_2O_4/Aerozine 50
(2) Designed by means of Rupe's velocity criteria for use with N_2O_4/UDMH
(3) Designed by means of Rupe's velocity criteria for use with N_2O_4/Aerozine 50
(4) Same as 2C-1 except oxidizer I.D. increased, fuel I.D. decreased to increase momentum ratio.
(5) Slight increase in momentum ratio: reduced L/D.
(6) Decreased impingement angle to 75°.
(7) Decreased impingement angle to 60°.
(8) Added 18 threads/in. over 3/4 of tube internal length to roughen surface.
(9) Same as 2C-8 except oxidizer orifices made elliptical, fuel orifice made square, to increase stream contact area.

TABLE 3

THEORETICAL PERFORMANCE OF N_2O_4/AEROZINE-50 AND N_2O_4/UDMH

M.R.	Oxidizer	Fuel	C^* ft/sec	I_{sp} sec	I_{sp} x sec	C_f	C_f x
1.0	N_2O_4	Aerozine 50[2]	5489	248.1	242.5	1.454	1.421
1.40	↓	↓	5721	260.0	254.2	1.462	1.430
1.80			5744	258.8	253.0	1.449	1.416
2.20			5610	254.4	248.8	1.459	1.427
2.60		Aerozine 50	5437	246.3	241.0	1.457	1.424
2.0		UDMH	5685	256.5	251.0	1.452	1.421
2.5		UDMH	5650	252.0	246.1	1.433	1.403
3.0	N_2O_4	UDMH	5505	245.0	239.5	1.430	1.398

(1) All performance calculations are based on shifting-equilibrium conditions, P_c = 500 psia, A_e/A_t = 3.0, $P_e = P_a$ = 14.7, and expansion half angle = 17.5°.

(2) Aerozine 50 is composed of 50 wt% of N_2H_4 and 50 wt% of UDMH.

TABLE 4

LARGE THRUST PER ELEMENT INJECTOR PERFORMANCE WITH N_2O_4/AEROZINE-50

TEST[4],[5] SERIES D-222-LJ-	F[6] (LB)	MIXTURE RATIO	P_c[1] (PSIA)	% OF P_c OSCILLATIONS	C^*[2] (FT/SEC)	% OF THEO C^*	I_{SP}[2] (SEC)	% OF THEO I_{SP}	C_F[2]	% OF THEO C_F	L^* (IN.)	L' (IN.)	A_T[3] ($IN.^2$)	TEST DURATION (SEC)	DATA POINT TIME (SEC)	REMARKS
1	8,928	2.05	706	± 10	5120	90.0	227	83.0	1.43	92.2	51	13	9.10	1.44	1.40	SATISFACTORY TEST; GAS LEAK ON INJECTOR FLANGE
2	9,064	1.81	720	± 7.5	5140	89.0	227	82.5	1.42	92.7	51	13	9.09	1.497	1.45	" " "
3	10,440	1.81	801	± 2.5	5730	99.3	260	94.5	1.46	95.3	100	25	9.09	1.502	1.46	SATISFACTORY TEST
4	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	MALFUNCTION DUE TO DISCONNECTED WIRE ON FUEL VALVE
5	8,290	1.84	743	± 2.5	5290	92.0	207	75.0	1.26	81.5	59	15	[illegible]	1.49	1.45	SATISFACTORY TEST
6	9,824	1.80	740	± 2.4	5180	89.8	242	88.1	1.50	98.1	59	15	[illegible]	1.52	1.47	" "
7	10,215	1.95	801	± 2.1	5700	99.6	256	93.0	1.45	93.4	100	25	9.02	1.50	1.50	" "
8	10,295	2.0	802	± 3.0	5700	99.9	258	93.8	1.46	93.9	100	25	9.0	1.50	1.50	" "
9	9,810	1.98	809	± 3.5	5750	100.4	246	89.5	1.38	89.3	100	25	8.96	1.50	1.50	" "
10	9,785	2.01	815	± 3.5	5800	101.7	246	89.5	1.37	88.0	100	25	8.96	1.50	1.50	" "
11	9,790	1.99	776	± 3.9	5930	103.8	246	89.5	1.34	86.3	100	25	9.67	3.58	3.55	SATISFACTORY TEST; NOZZLE THROAT ERODED

NOTES:
(1) P_c IS MEASURED ON THE INJECTOR FACE AND CORRECTED TO PLENUM BY DIVIDING BY 1.01
(2) C^*, I_{SP} AND C_F VALUES ARE CORRECTED TO 800 PSIA PLENUM CHAMBER PRESSURE TO FACILITATE COMPARISON WITH THEORETICAL
(3) BASED ON FINAL THROAT AREA
(4) TESTS -1 AND -2 WITH THE FOLLOWING FUEL COMPOSITION: UDMH - 46.6%; N_2H_4 - 51.1%; H_2O - 2.3%
(5) TESTS -3 THRU -11 WITH THE FOLLOWING FUEL COMPOSITION: UDMH - 48.5%; N_2H_4 - 50.3%; H_2O - 1.2%
(6) CHAMBER CONTRACTION RATIO, A_c/A_T = 4.0; NOZZLE EXPANSION RATIO, A_E/A_T = 4.53

TABLE 5

TEST DATA LARGE THRUST/ELEMENT COMBUSTION STABILITY
22.5-IN. DIA PULSE MOTOR PROPELLANTS N_2O_4 - AEROZINE-50

Test No. D-223-LJ	Spray Fan Orientation (4)	P_c (PSIA)	Mixture Ratio	W_o (LB/SEC)	W_f (LB/SEC)	ΔP_o (PSI)	ΔP_f (PSI)	Pulse V-10	V-15	V-20	V-40	V-80	Stability Index(3)	C^* FT/SEC	Remarks
1	PARALLEL	962	1.97	29.86	15.16	74	52	S	S	S(1)	-	-	1.40	6500	CONTINUOUS LOW AMPLITUDE INSTABILITY AFTER FIRST PULSE
2	↑	444	1.95	29.70	15.25	73	52	SPONTANEOUS INSTABILITY					0	5500	WITH WINDOW, MULTI-FRONT INSTABILITY
3	|	433	1.97	29.91	15.15	75	52	S	S	U(2)	-	-	1.55	5380	WITH WINDOW, VALVE OPENING TIME REDUCED ONE-HALF, FILM UNDEREXPOSED
4	|	651	1.96	29.56	15.08	72	50	S	S	U	-	-	1.15	5910	WITH WINDOW, MULTI-FRONT INSTABILITY
5	|	625	1.95	29.70	15.22	73	52	-	S	U	-	-	1.20	5630	WITH WINDOW, SINGLE FRONT INSTABILITY
6	|	823	1.96	29.72	15.17	73	52	-	S	U	-	-	.98	5580	
7	|	915	2.12	29.62	13.94	72	42	-	S	U	-	-	.90	5770	
8	|	-	-	-	-	-	-	SPONTANEOUS INSTABILITY					0	-	PREMATURE SHUTDOWN BY MALFUNCTION TIMER
9	|	-	-	-	-	-	-	SPONTANEOUS INSTABILITY					0	-	INSUFFICIENT TIME INTERVAL FOR DATA REDUCTION
10	|	783	1.55	19.49	12.57	32	35	SPONTANEOUS INSTABILITY					0	5750	
11	|	795	1.57	28.48	18.17	68	71	SPONTANEOUS INSTABILITY					0	5870	
12	↓	872	2.39	36.43	15.27	110	52	-	S	U	-	-	.94	5800	
13	PARALLEL	876	1.99	36.40	18.26	110	72	-	S	U	-	-	.94	6170	
14	PERPENDICULAR	862	2.00	36.86	18.43	112	73	-	S	S	-	-	1.52	5824	
15	↑	874	2.39	37.15	15.54	115	52	-	S	S	-	-	1.51	5656	
16	|	-	-	-	-	-	-	-	-	-	-	-	-	-	PREMATURE SHUTDOWN BY MALFUNCTION TIMER (SET TO CUT OFF .10 SEC TOO SOON) INSUFFICIENT TIME INTERVAL FOR DATA REDUCTION
17	|	799	1.59	20.44	12.88	35	36	-	S	S	-	-	1.62	5620	
18	|	802	1.60	20.47	12.77	35	36	-	-	S	S	U	3.50	5660	
19	|	852	2.45	30.49	12.45	78	34	-	S	S	S	-	3.41	5450	
20	|	798	2.04	37.77	18.56	118	75	-	-	S	S	U	3.8	5620	
21	|	861	2.42	37.58	15.51	117	53	-	-	S	S	S	-	5480	
22	↓	852	1.95	29.72	15.21	74	50	-	-	U	-	-	.96	5840	
23	PERPENDICULAR	844	1.60	28.90	18.03	70	70	-	-	U	-	-	.96	6080	OXIDIZER ELEMENT TUBES ERODED ON SIDE FACING INDUCED TANGENTIAL INSTABILITY WAVE

NOTES:
(1) S - STABLE
(2) U - UNSTABLE
(3) FOR LARGEST STABLE PULSE
(4) INJECTOR P/N 073383, ORIENTATION WITH RESPECT TO CHAMBER WALL

TABLE 6

AEROZINE-50 COMPOSITION

Test Number	% N_2H_4	% UDMH	% H_2O
D-223-LJ-1, -2	51.6	48.9	-
-3	51.1	49.3	=
-4 to -6	51.3	47.9	0.8
-7 to -9	51.2	49.0	=
-10 to -19	51.0	49.8	=
-20 to -23	48.8	50.6	0.6
Spec Requirements	50.2 min.	47.0 min.	1.8 max.

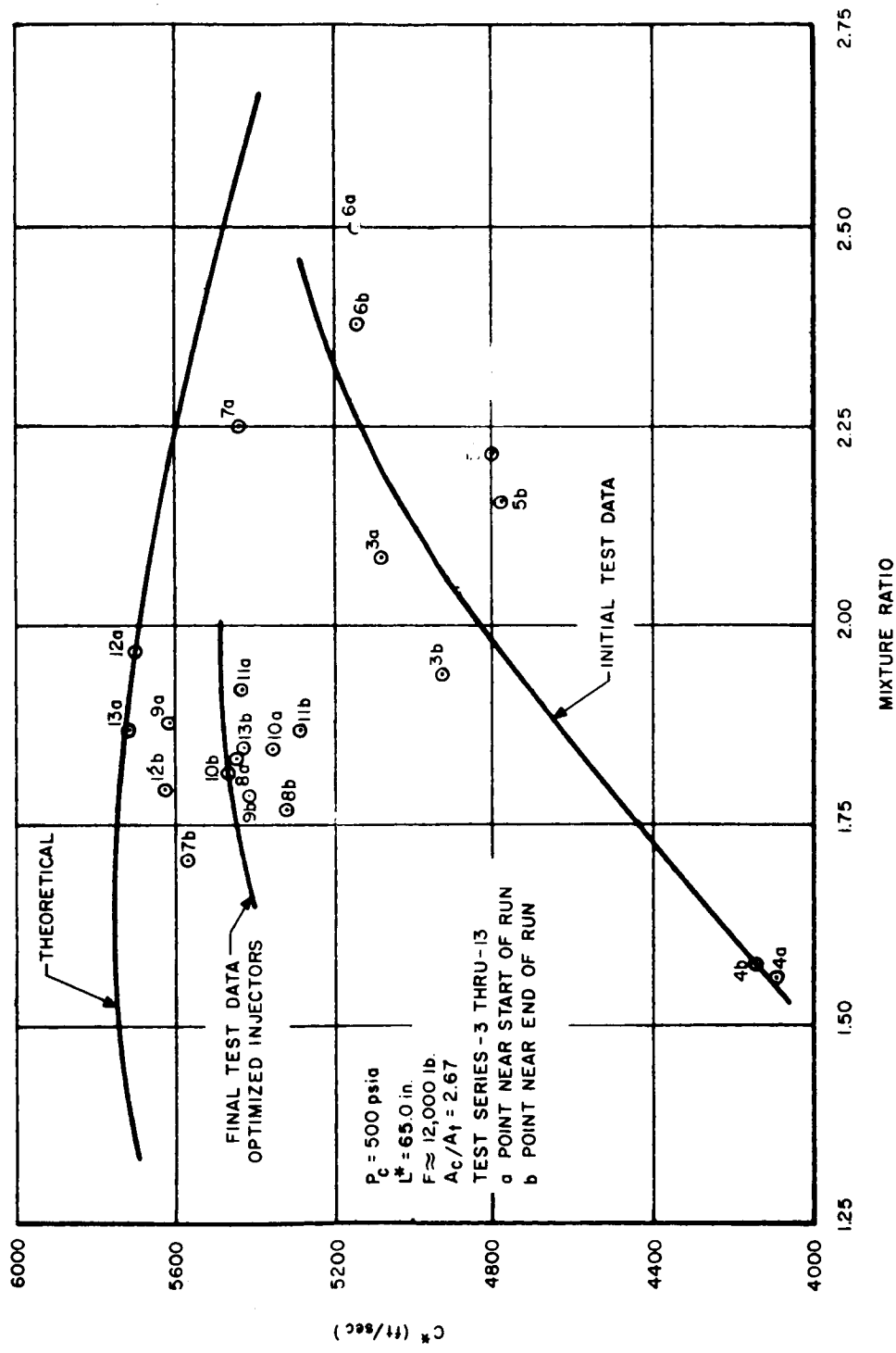
TABLE 7

PHASE III RESULTS OF LARGE THRUST PER ELEMENT INJECTOR PROGRAM

(6) Test No. D-225-LJ-	F (LB)	Mixture Ratio	(1)(2) P_c (PSIA)	% of P_c Oscillations	$\dot{w}_T$ (LB/SEC)	C^* (FT/SEC)	% of Theor. C	(3) I_{sp} (SEC)	(4) % of Theor. I_{sp}	C_F	L^* (IN.)	L (IN.)	A_{T_2} (IN.^{2})	Test Duration (SEC)	Data Point Time (SEC)	(5) Element Orientation	Test Date	Remarks	
-1	10,600	2.09	789 (1)	± 4.5	40.8	5722	100.6	260.1	97.0	1.46	59	13	9.18	.83	.71	R	4-18-61	Satisfactory Test	
-2	10,640	1.90	809 (2)	± 3.0	42.8	5613	97.7	248.7	92.5	1.43	59	13	9.22	1.5	1.43	R	4-19-61	"	"
-3	10,680	1.90	793 (2)	± 4.0	42.8	5510	95.9	249.4	93.1	1.46	59	13	9.24	1.?	1.18	R	4-20-61	"	"
-4	10,850	1.89	820 (1)	± 4.5	43.1	5619	97.7	251.5	93.5	1.44	59	13	9.18	1.3	1.20	R	4-20-61	"	"
-5	10,416	1.89	802 (1)	± 4.0	42.3	5603	97.4	246.2	91.6	1.41	59	13	9.18	1.3	1.21	T	4-21-61	"	"
-6	10,376	1.88	811 (2)	± 3.5	42.9	5589	97.2	241.9	90.0	1.39	59	13	9.18	1.3	1.20	T	4-24-61	"	"
-7	10,696	1.92	829 (1)	± 4.0	42.8	5731	99.8	250.1	93.0	1.41	59	13	9.18	1.3	1.20	45	4-24-61	"	"
-8	10,816	1.91	831 (2)	± 3.4	43.1	5697	99.2	250.7	93.2	1.42	100	23	9.19	1.3	1.23	R	4-25-61	"	"
-9	10,672	1.92	835 (1)	± 4.2	43.1	5738	99.9	247.8	92.1	1.39	51	11	9.19	1.3	1.22	R	4-26-61	"	"
-10	10,528	1.90	830 (1)	± 3.0	42.5	5778	100.5	247.7	92.1	1.38	51	11	9.19	1.4	1.30	45	4-28-61	"	"

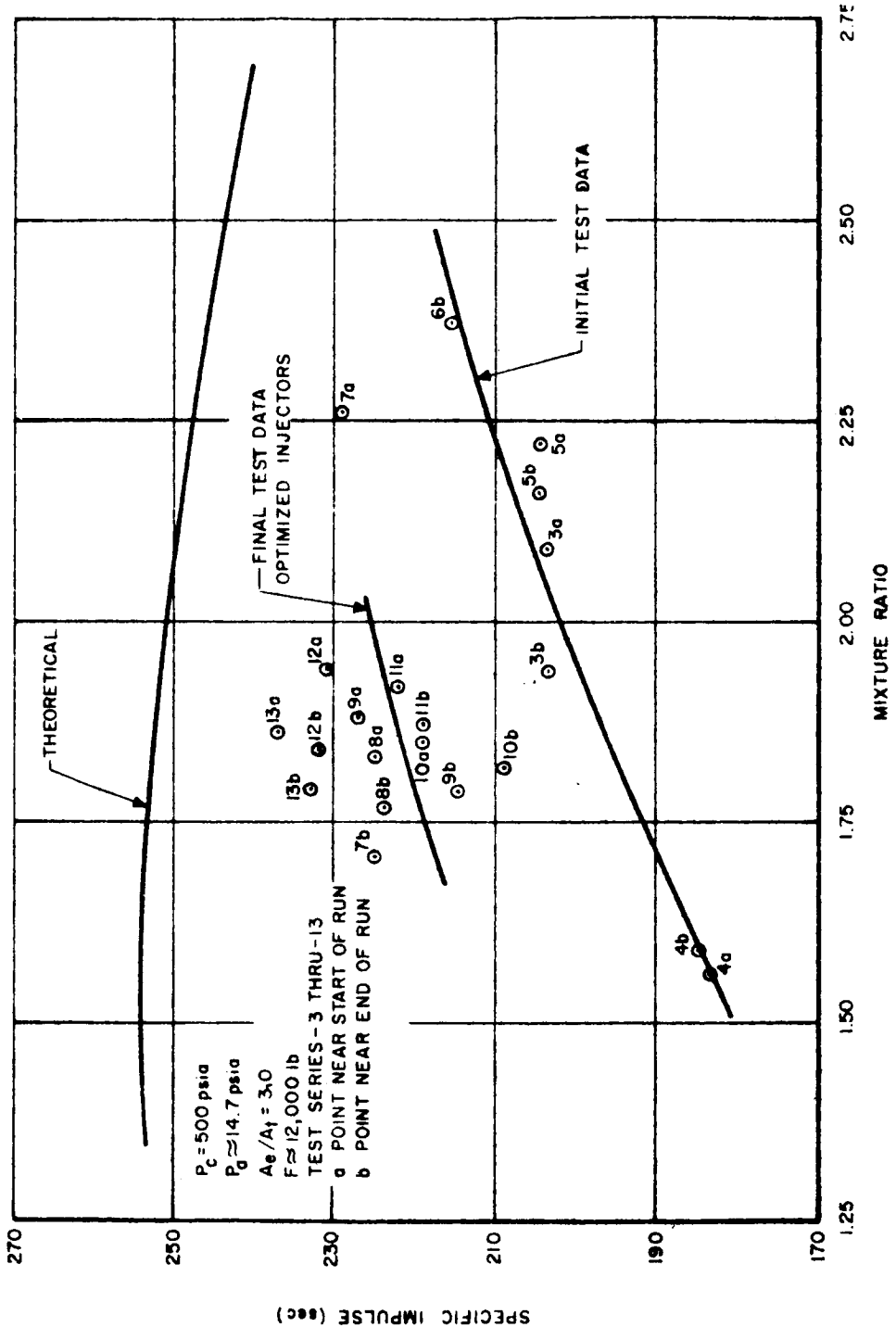
NOTES:
(1) P_c IS MEASURED AT THE ENTRANCE TO THE NOZZLE AND CORRECTED TO PLENUM BY DIVIDING BY .987.
(2) P_c IS MEASURED AT THE INJECTOR FACE AND CORRECTED TO PLENUM BY DIVIDING BY 1.01.
(3) CHAMBER CONTRACTION RATIO A_C/A_T = 4.0; NOZZLE RATIO, A_E/A_T = 4.53.
(4) THEORETICAL I_{SP} IS CORRECTED FOR DIVERGENCE ANGLE BY MULTIPLYING BY .982.
(5) R - SPRAY FAN RADIAL TO CHAMBER WALL; T - SPRAY FAN TANGENTIAL TO CHAMBER WALL; 45 - SPRAY FAN 45° TO CHAMBER WALL.
(6) ALL TESTS WERE CONDUCTED WITH N_2O_4/AEROZINE-50.

CONFIDENTIAL



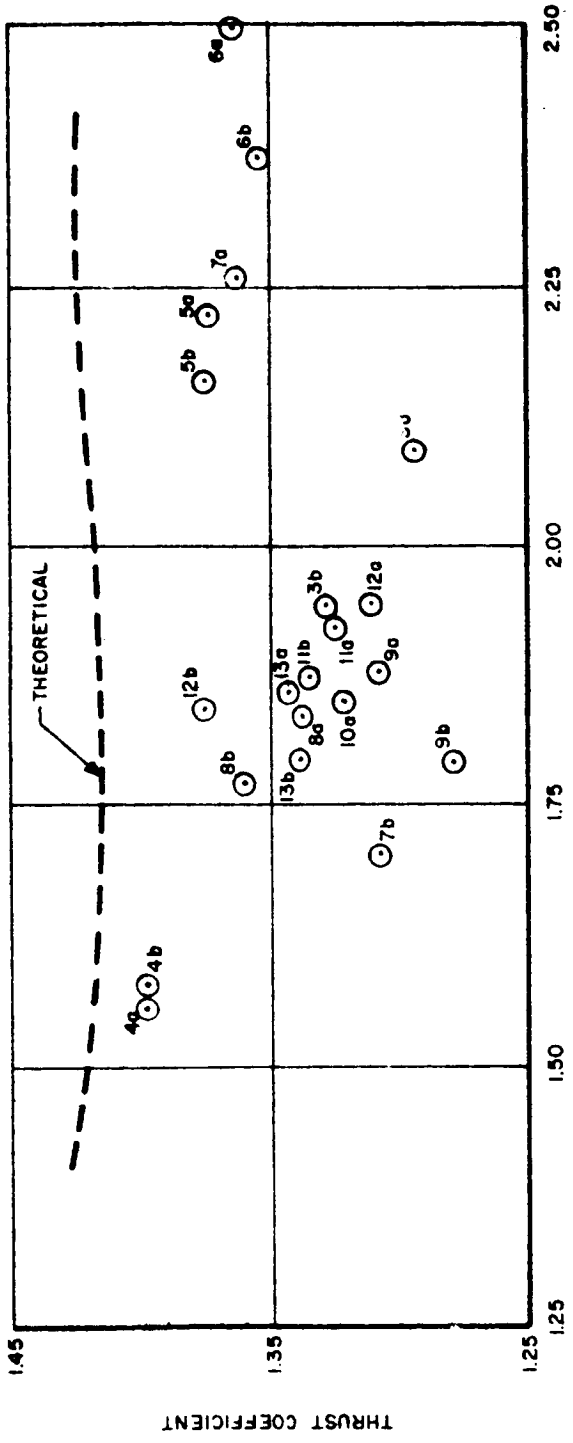
Characteristic Exhaust Velocity vs Mixture Ratio for N_2O_4/Aerozine-50

Figure 1



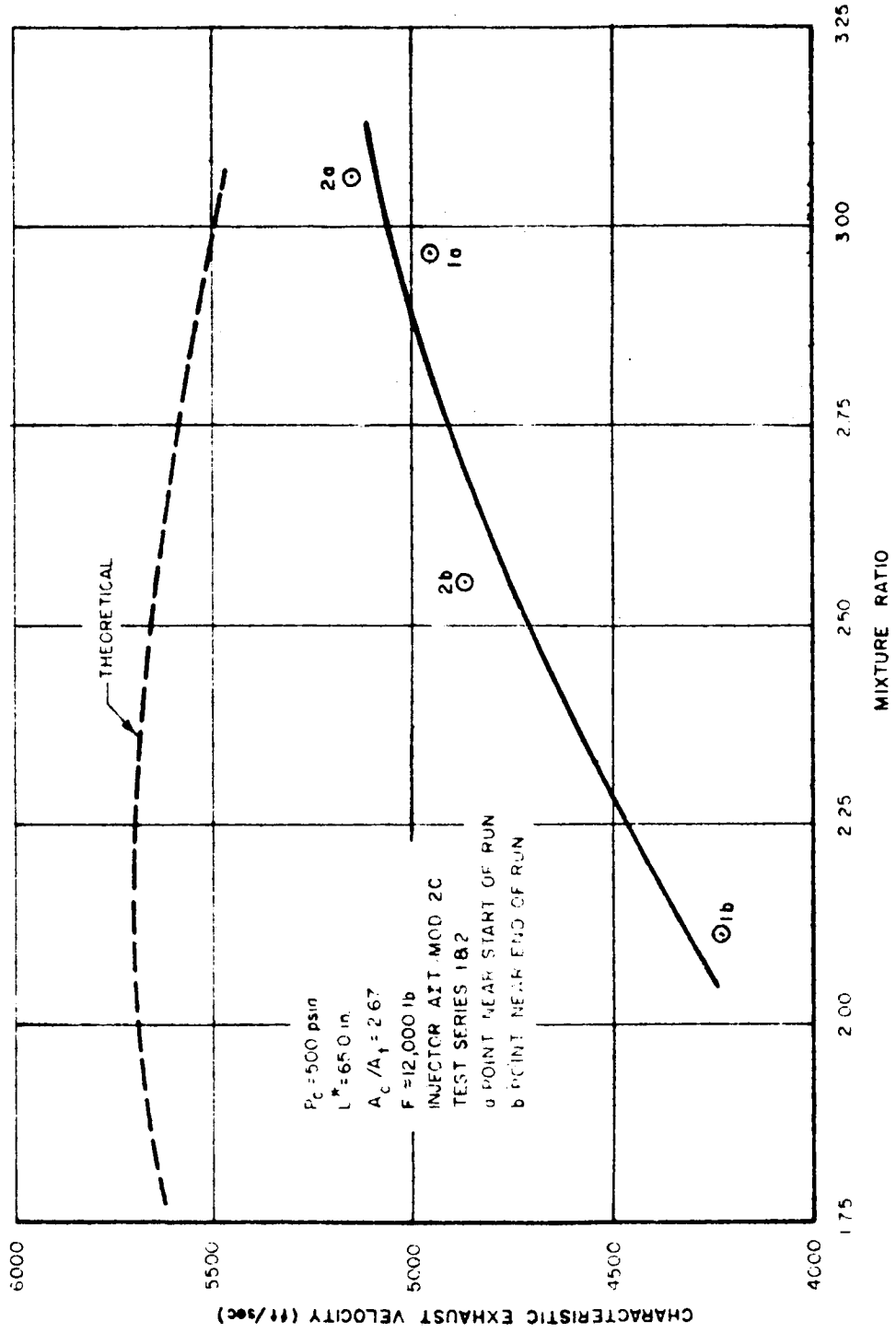
Specific Impulse vs Mixture Ratio for N_2O_4/Aerozine-50

Figure 2



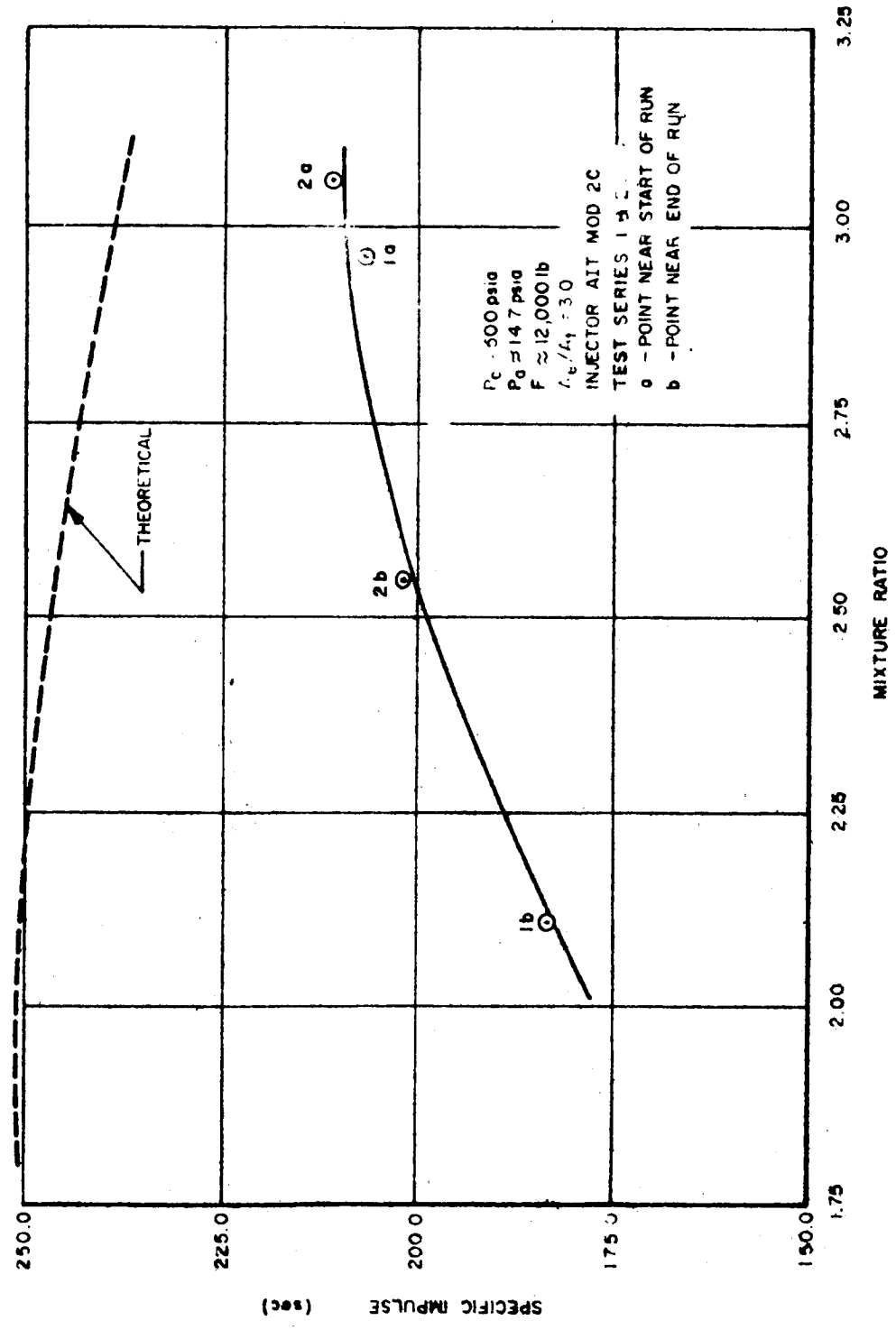
Thrust Coefficient vs Mixture Ratio for N_2O_4/Aerozine-50

Figure 3



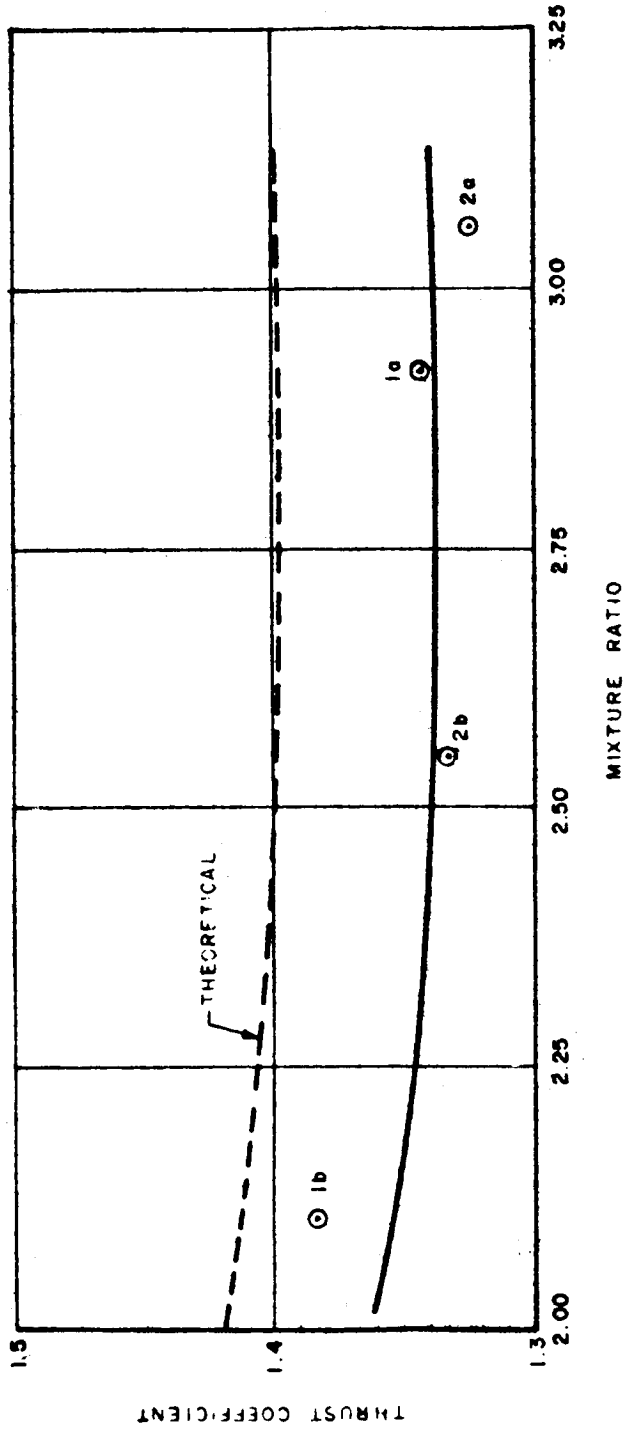
Characteristic Exhaust Velocity vs Mixture Ratio for N_2O_4/UDMH

Figure 4



Specific Impulse vs Mixture Ratio for N_2O_4/UDMH

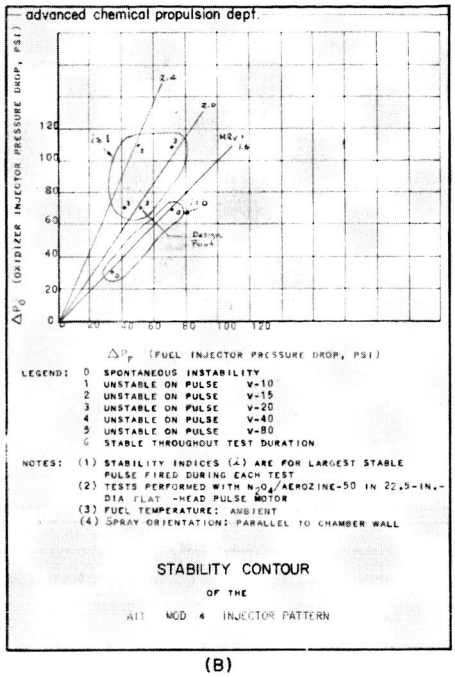
Figure 5



Thrust Coefficient vs Mixture Ratio for N_2O_4/UDMH

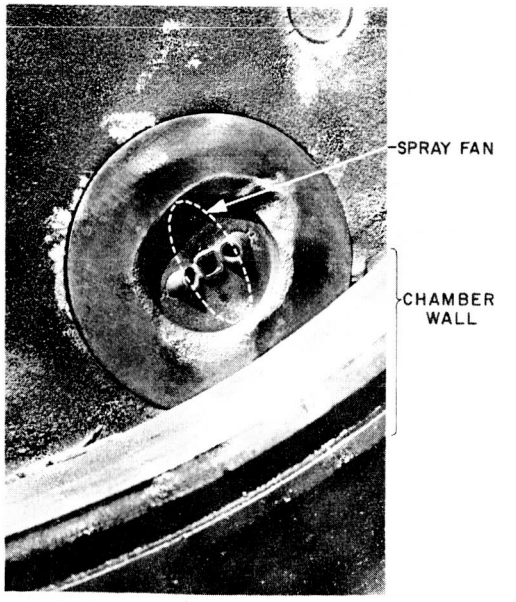
Figure 6

(A)

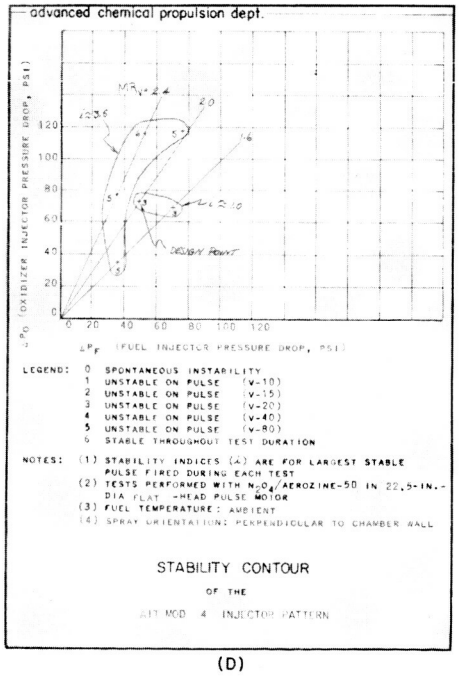


(B)

INJECTOR ELEMENT
SPRAY FAN PARALLEL TO WALL



(C)

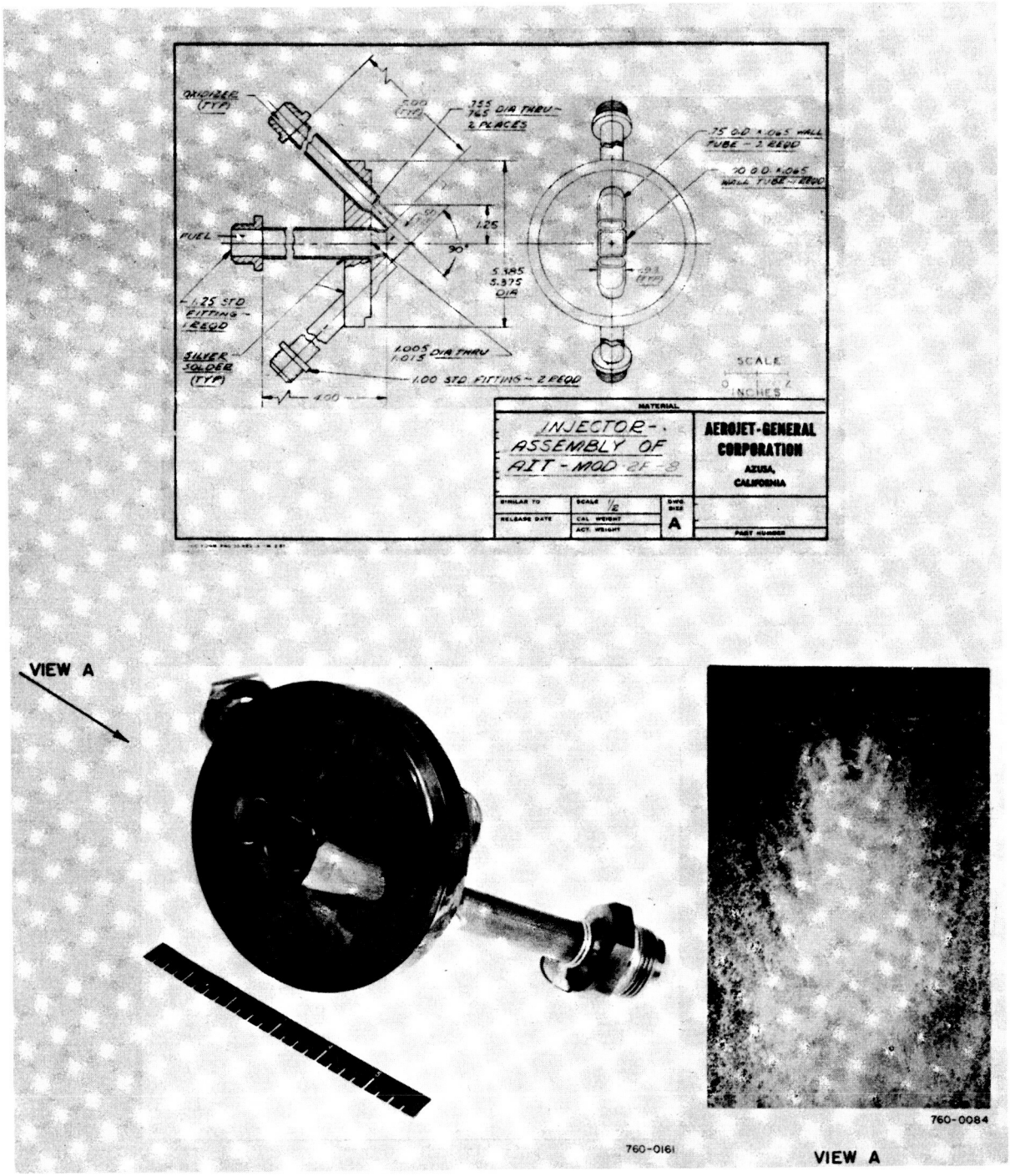


(D)

INJECTOR ELEMENT
SPRAY FAN PERPENDICULAR TO WALL

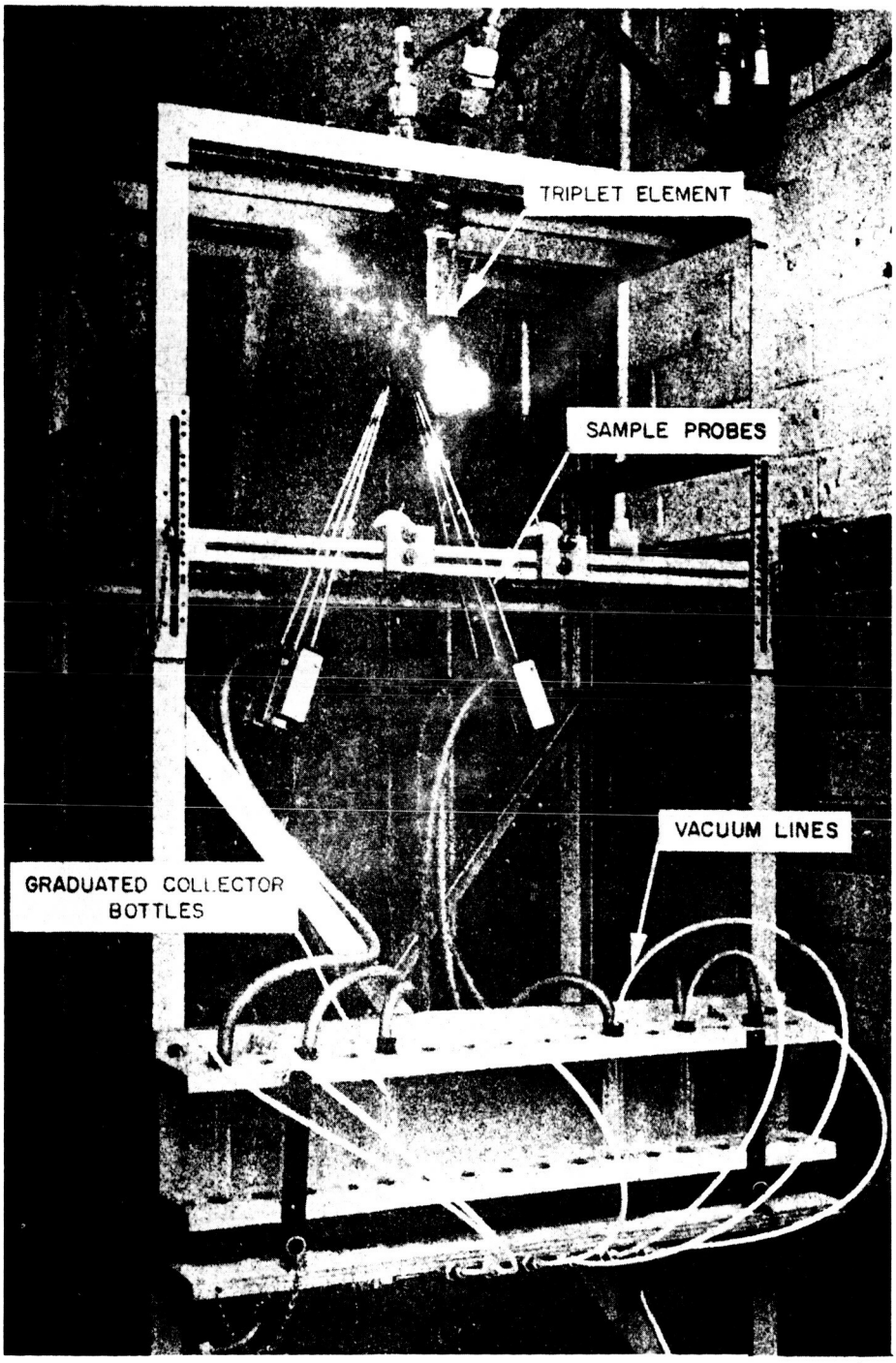
Injector Element Orientations and Stability Contours
for Large Thrust per Element Triplet Injector

Figure 9



Design and Spray Characteristics AIT Mod 2F-8 Injector Element

Figure 10



Nonreactive Spray Test Installation

Figure 11

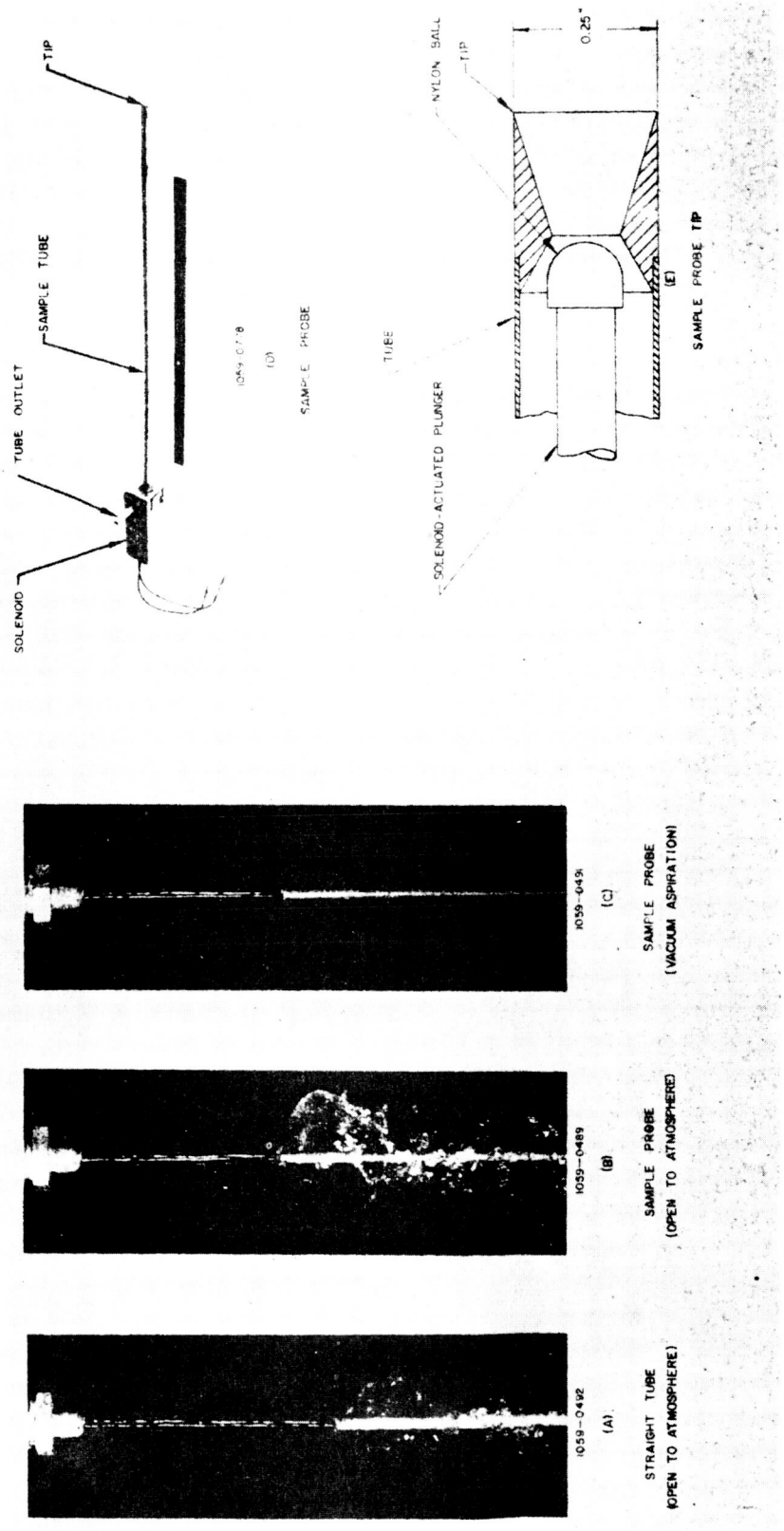
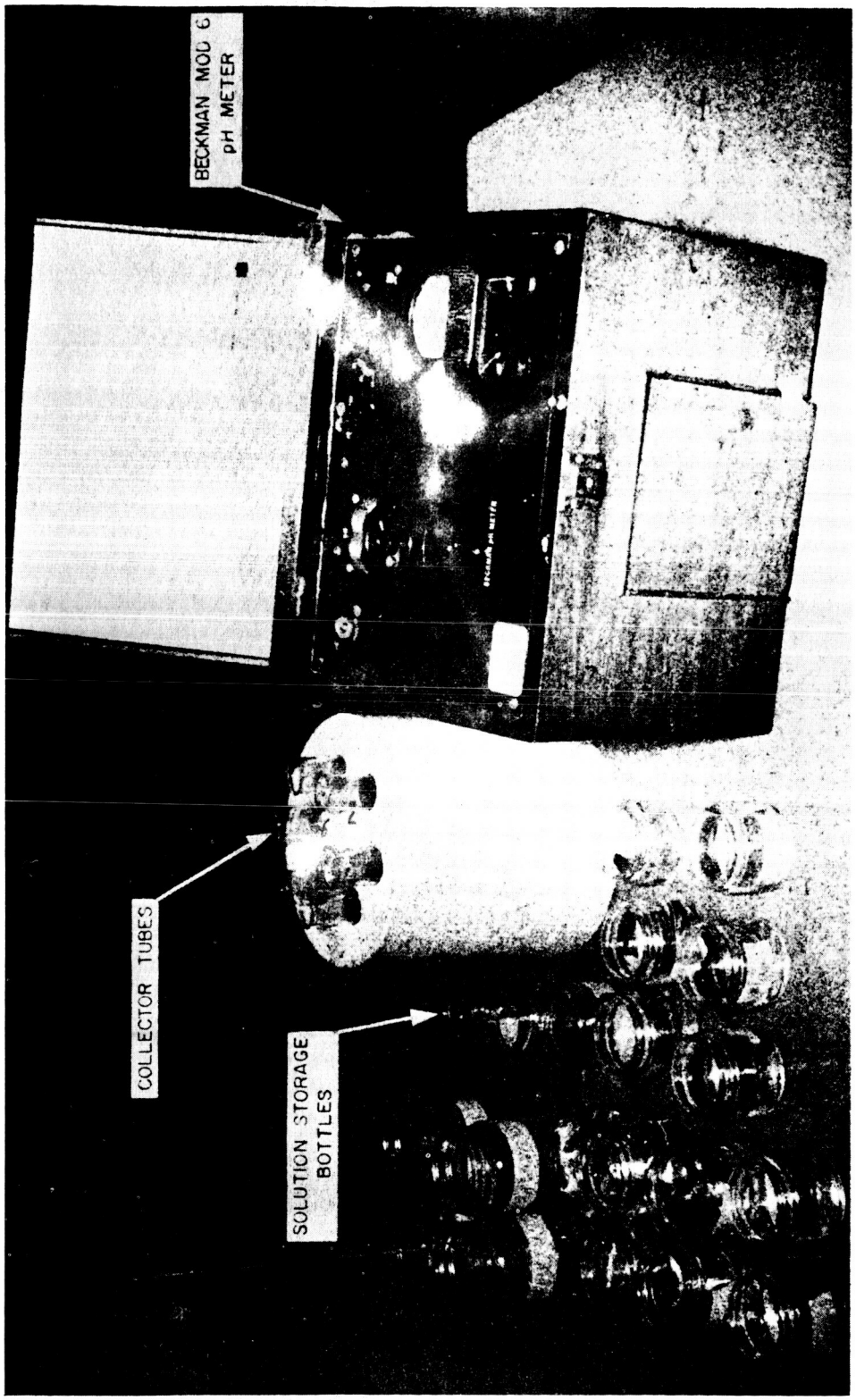
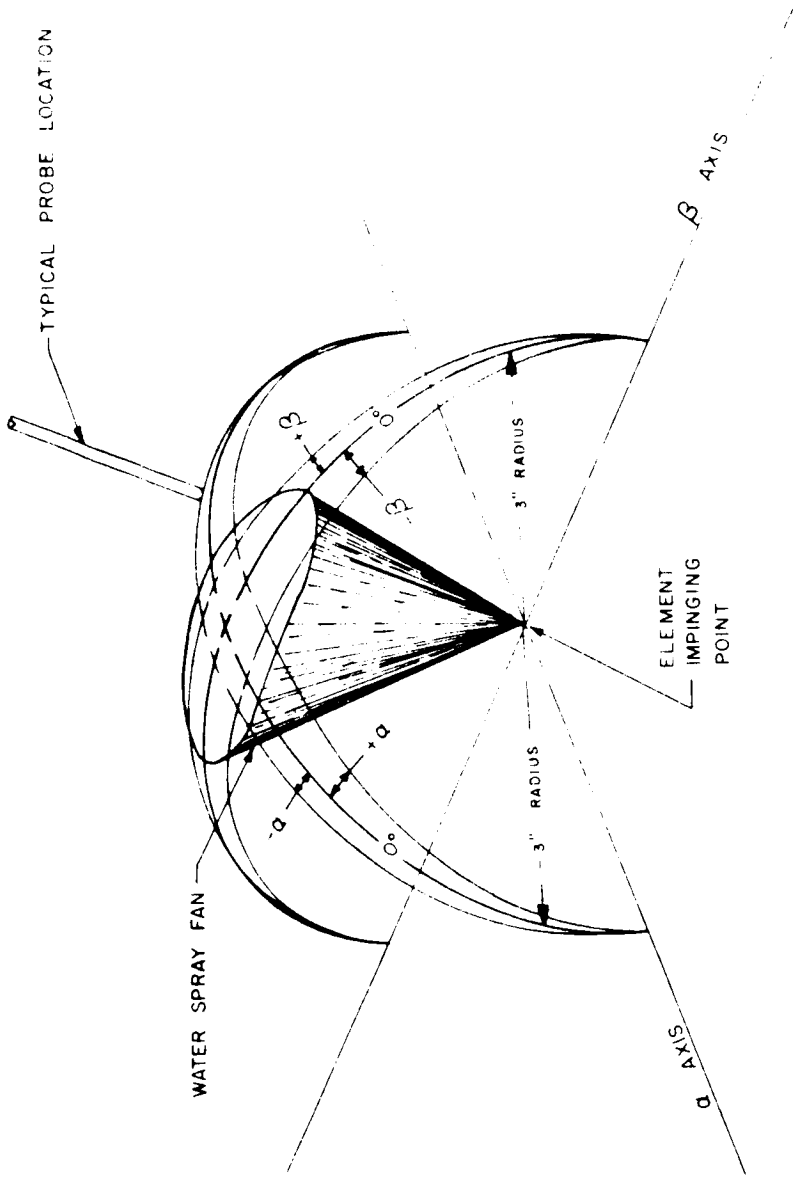

TIP
SAMPLE TUBE
TUBE OUTLET
SOLENOID
(D)
SAMPLE PROBE
TUBE
NYLON BALL
TIP
0.25"
SOLENOID-ACTUATED PLUNGER
(E)
SAMPLE PROBE TIP
1059-0491
(C)
SAMPLE PROBE
(VACUUM ASPIRATION)
1059-0489
(B)
SAMPLE PROBE
(OPEN TO ATMOSPHERE)
1059-0492
(A)
STRAIGHT TUBE
(OPEN TO ATMOSPHERE)

Figure 12



Mixture Ratio Sample Analysis Equipment

Figure 13

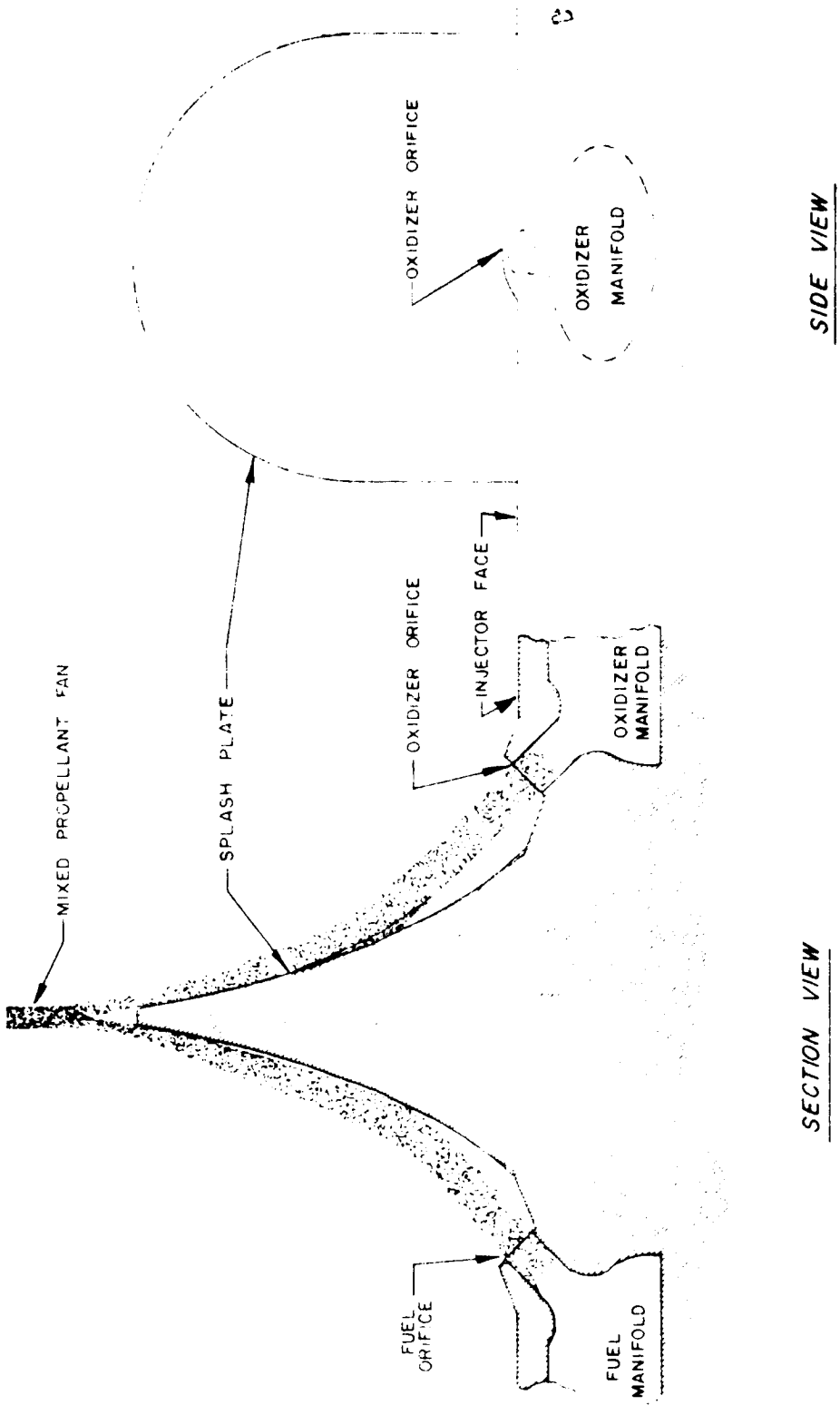


± α = ANGLE OF ROTATION AROUND α AXIS

± β = ANGLE OF ROTATION AROUND β AXIS

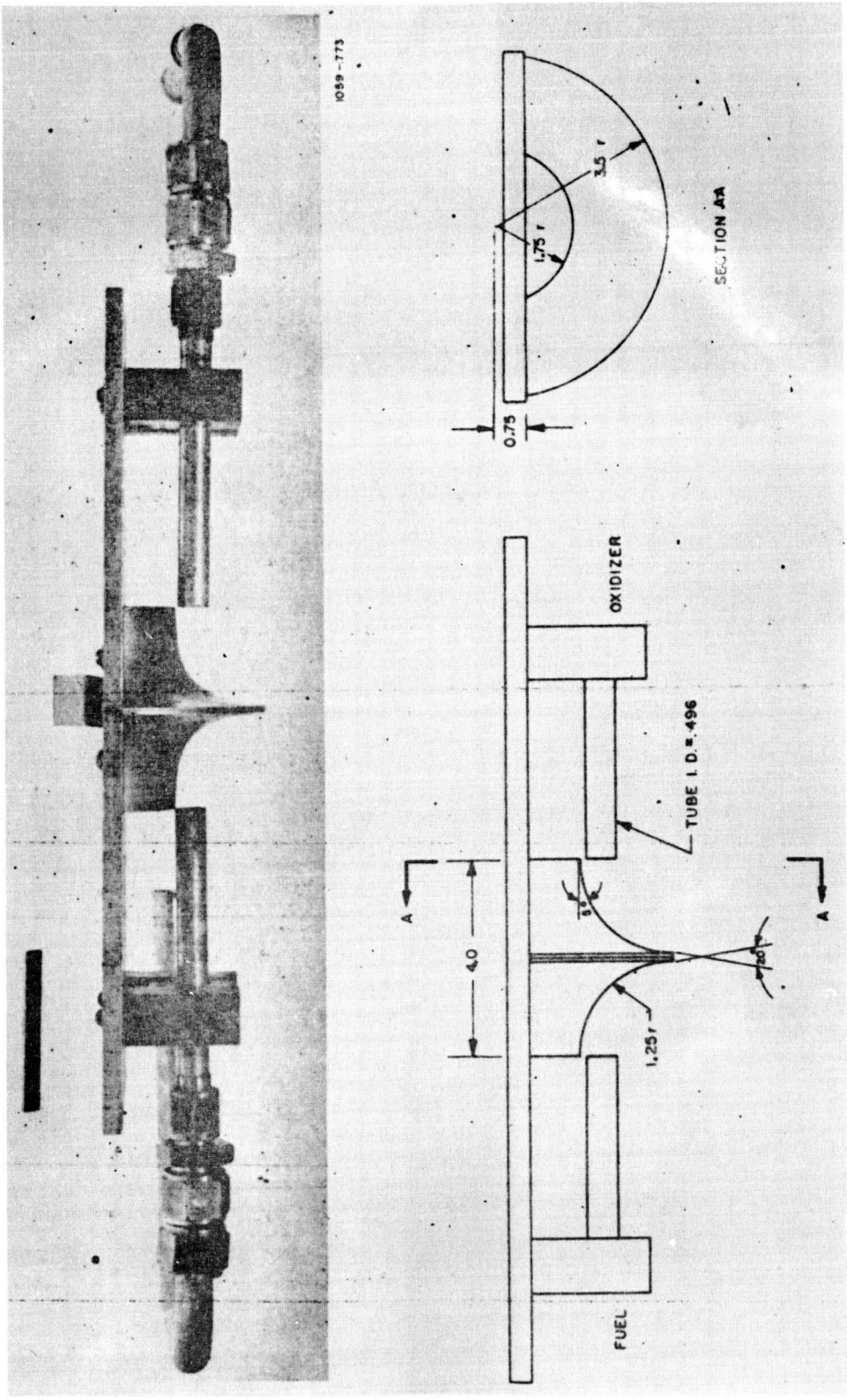
Orientation of Sample Probes

Figure 14



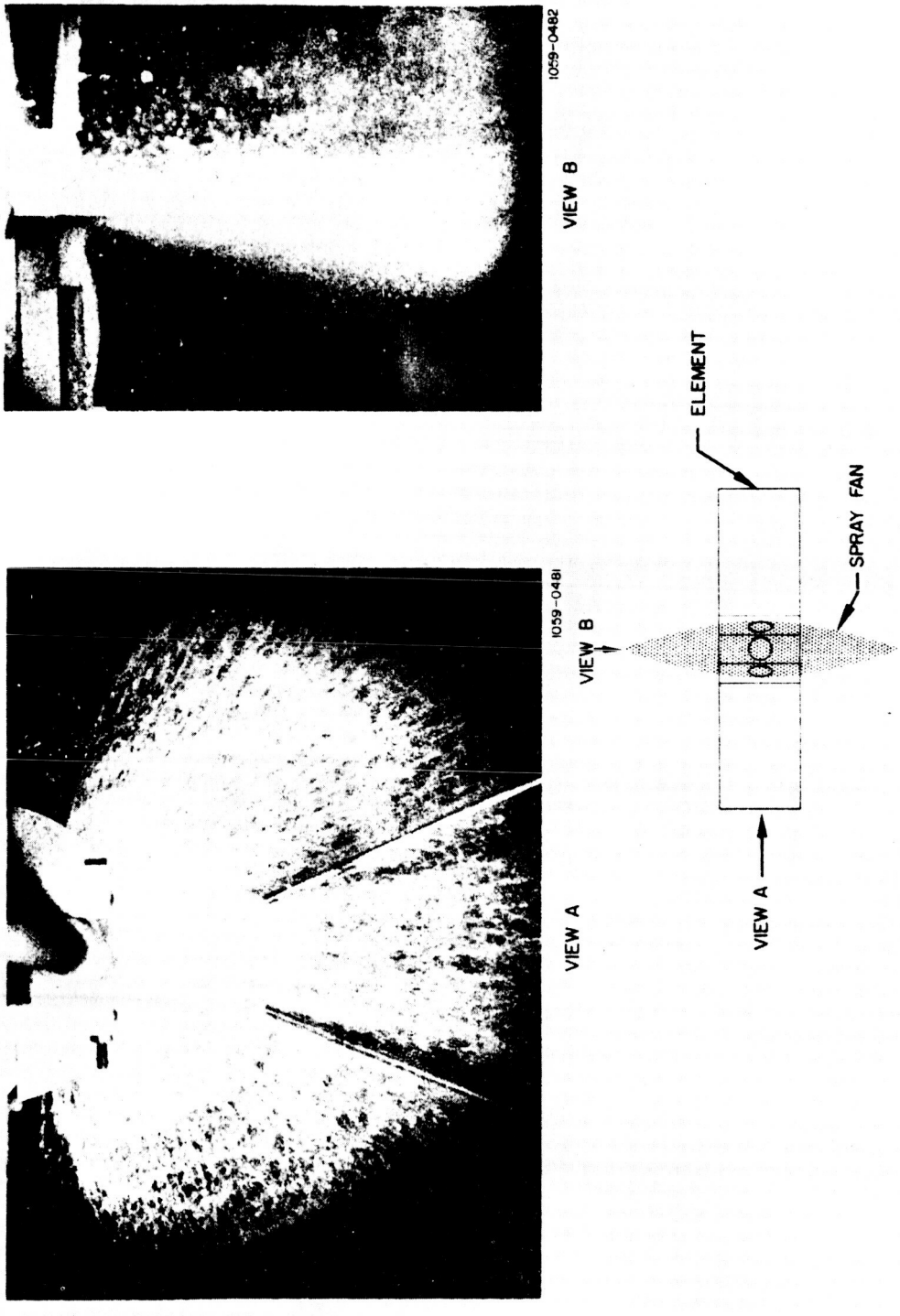
Large Thrust Injection Element (5,000 lb)

Figure 15



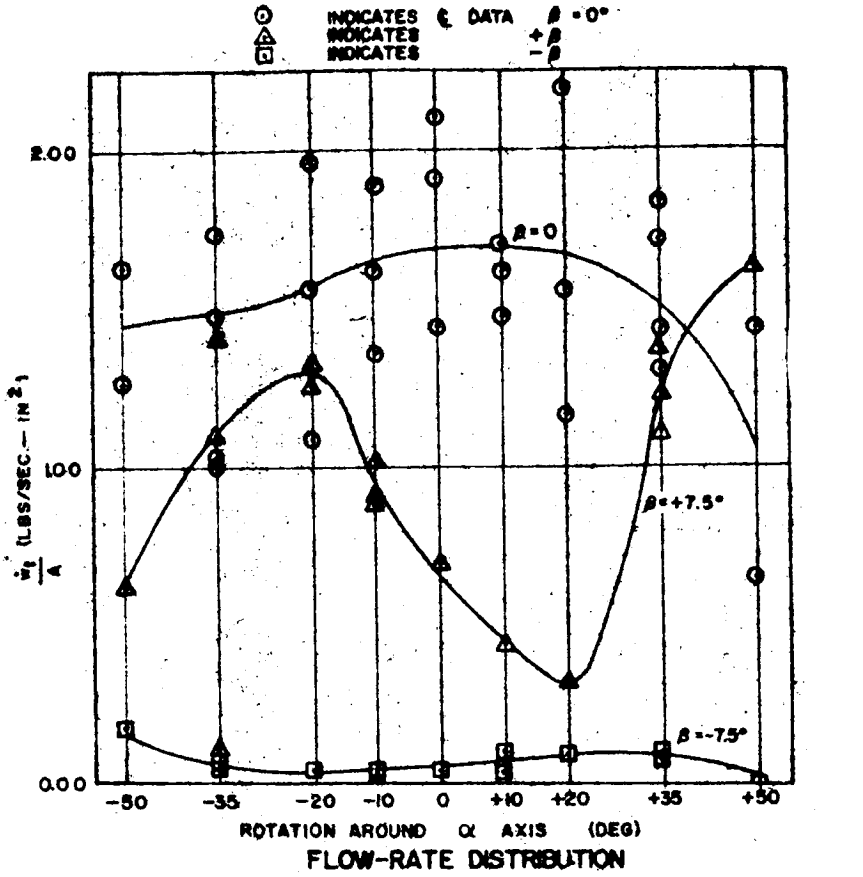
Splash-Plate Element AIT Mod 1

Figure 16

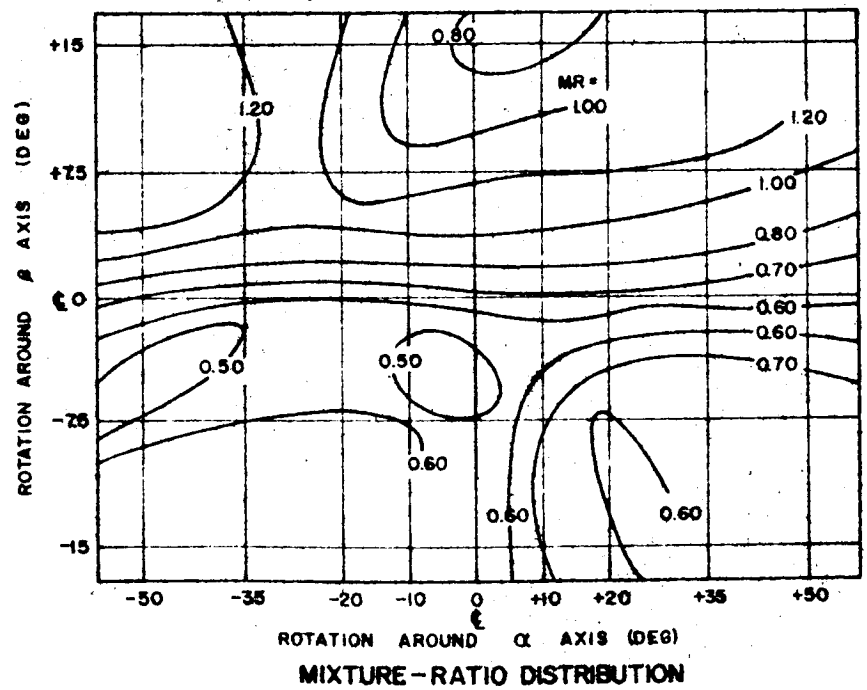


Splash Plate Spray Pattern AIT Mod 1

Figure 17



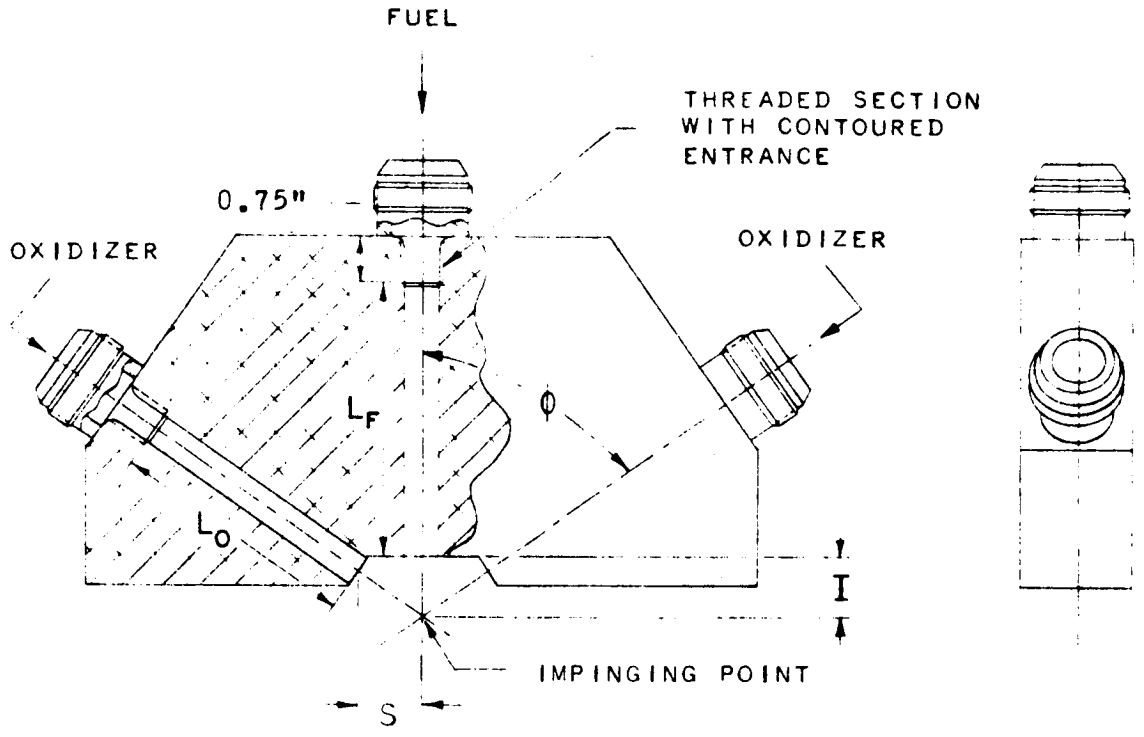
FLOW-RATE DISTRIBUTION



MIXTURE-RATIO DISTRIBUTION

Flow-Rate and Mixture-Ratio Distribution AIT Mod 1 (Splash Plate) Element

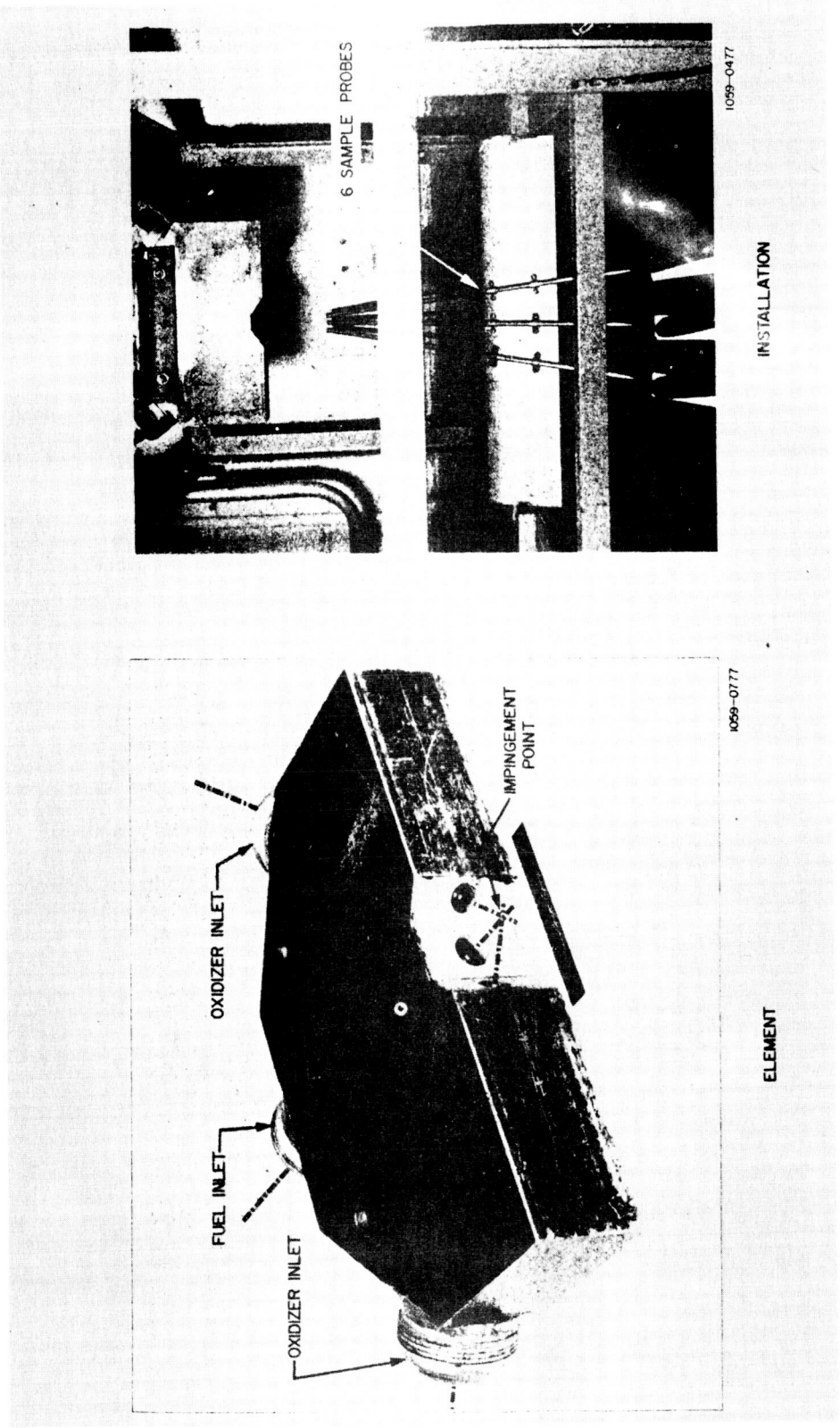
Figure 18



PATTERN DESIGNATION	OXIDIZER ORIFICE DIA D_O (IN.)	FUEL ORIFICE DIA D_F (IN.)	ϕ (DEG)	S (IN.)	I (IN.)	L_O/D_O	L_F/D_F
AIT MOD 2	0.478	0.533	45	0.750	1.000	8.9	8.8
AIT MOD 2A	0.478	0.533	60	1.200	1.000	9.1	8.8
AIT MOD 2B	0.478	0.533	35	0.625	1.100	9.8	8.8

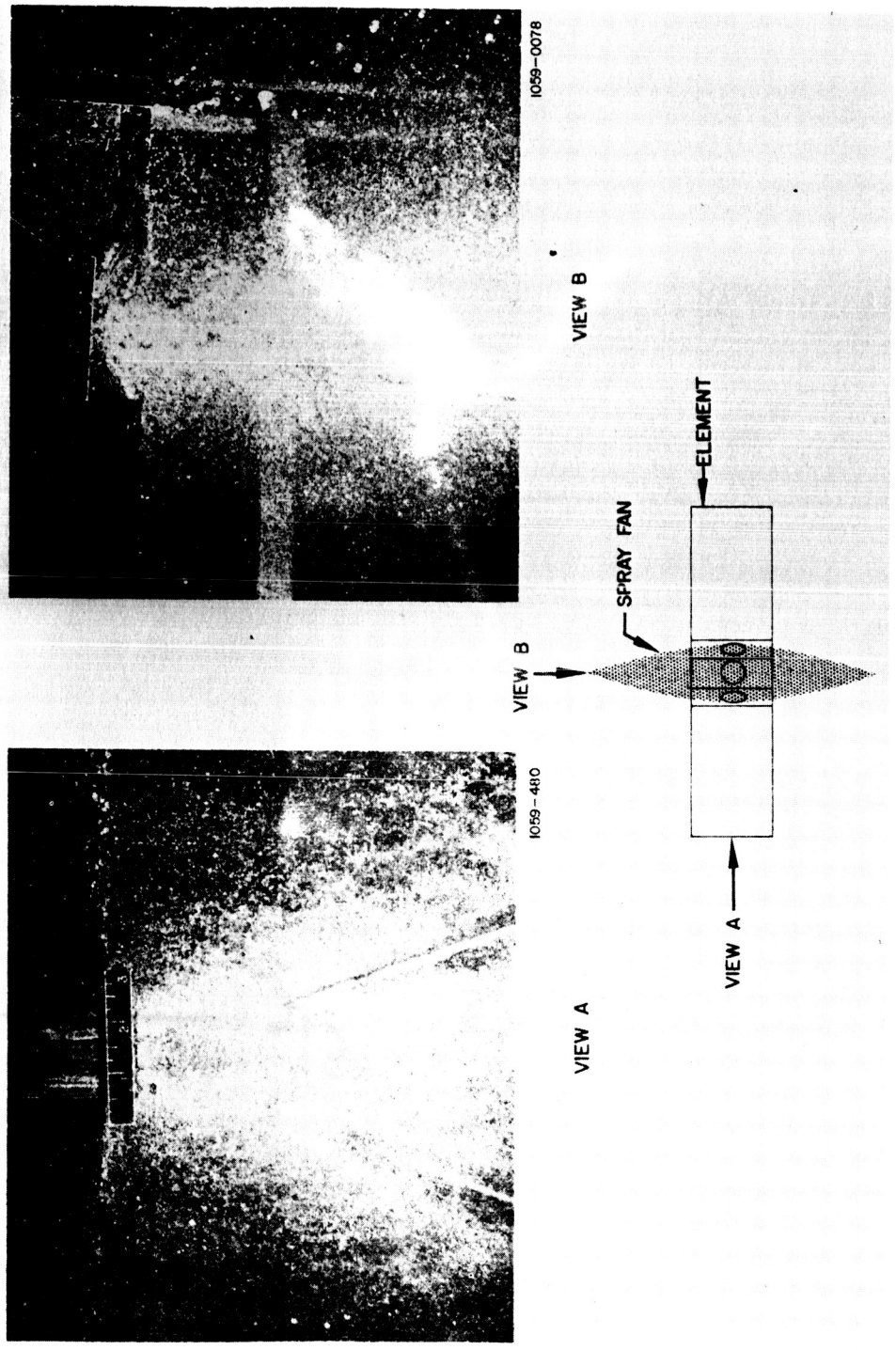
Triplet Element, AIT Mod 2 Type

Figure 19



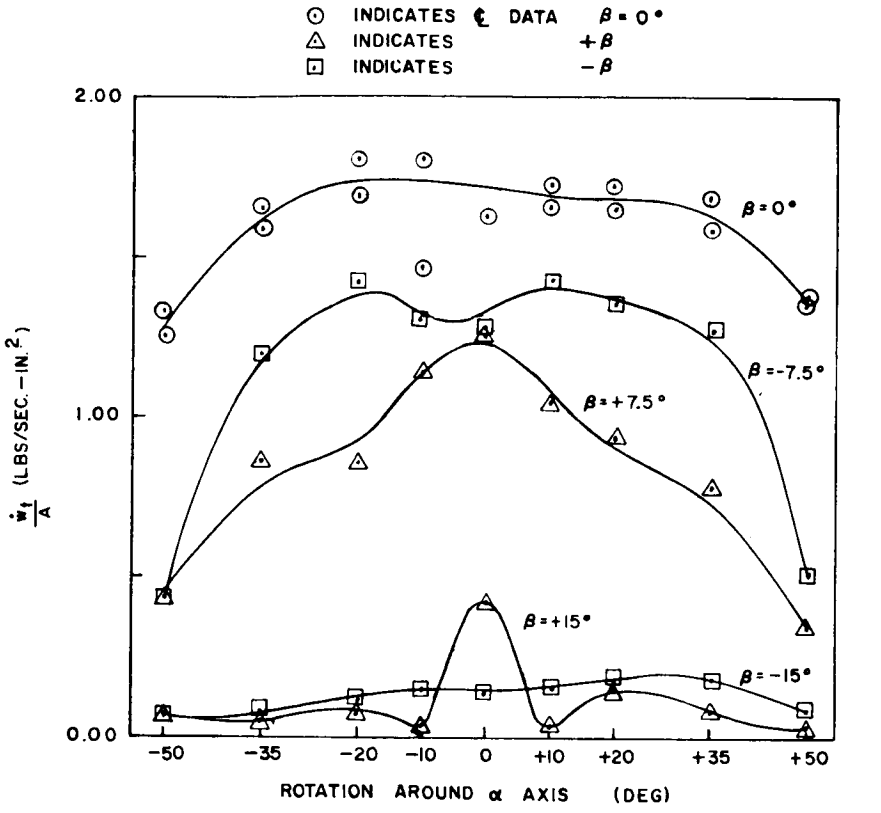
Typical Triplet Element Installation

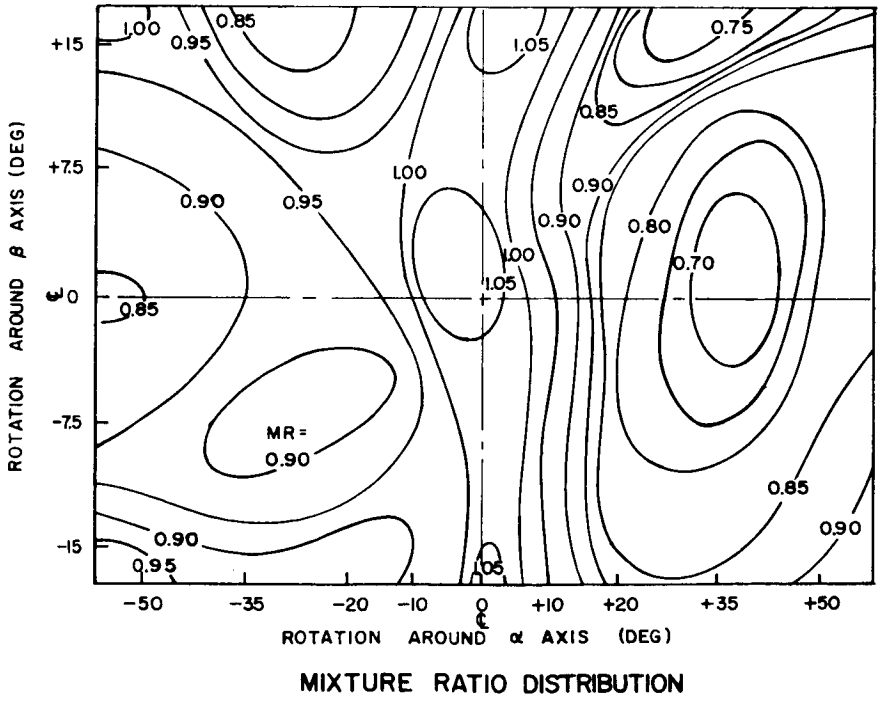
Figure 20



Triplet Spray Pattern AIT Mod 2

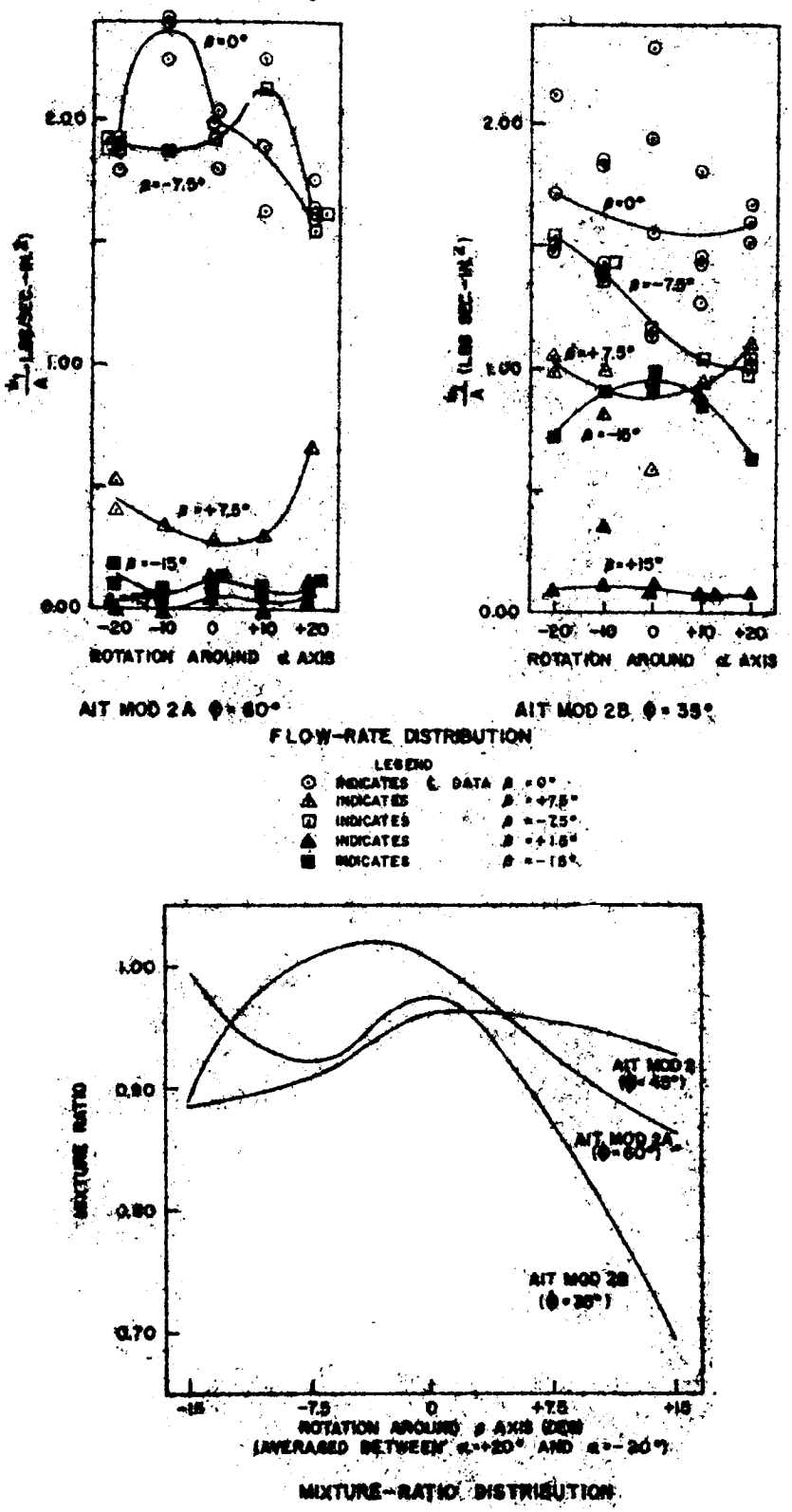
Figure 21





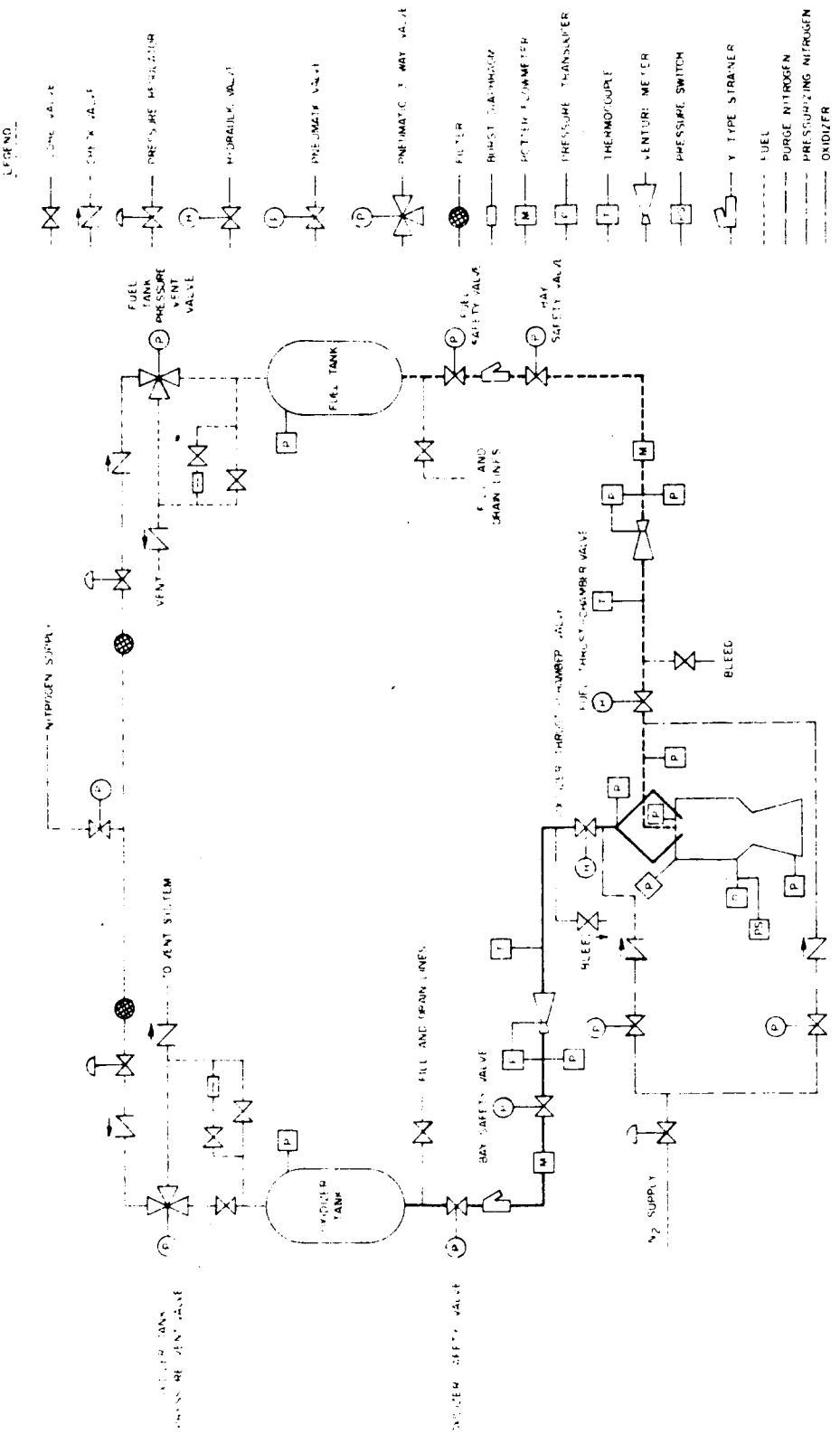
Flow-Rate and Mixture-Ratio Distributions for AIT Mod (ϕ = 45°) (Triplet) Element

Figure 22



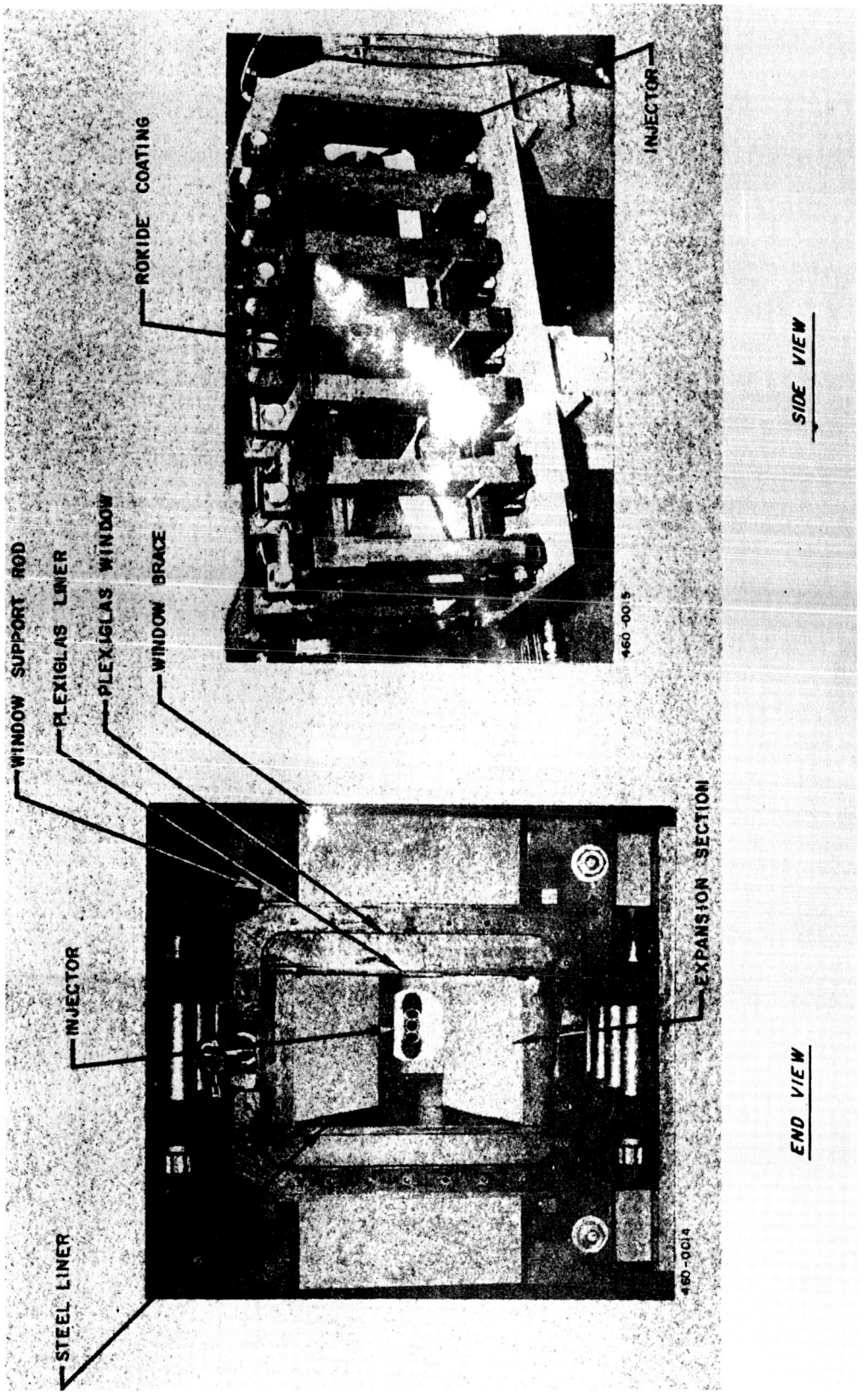
Flow-Rate and Mixture-Ratio Distributions for AIT Mod 2A and 2B (Triplet) Elements (ϕ = 60° and 35°)

Figure 23



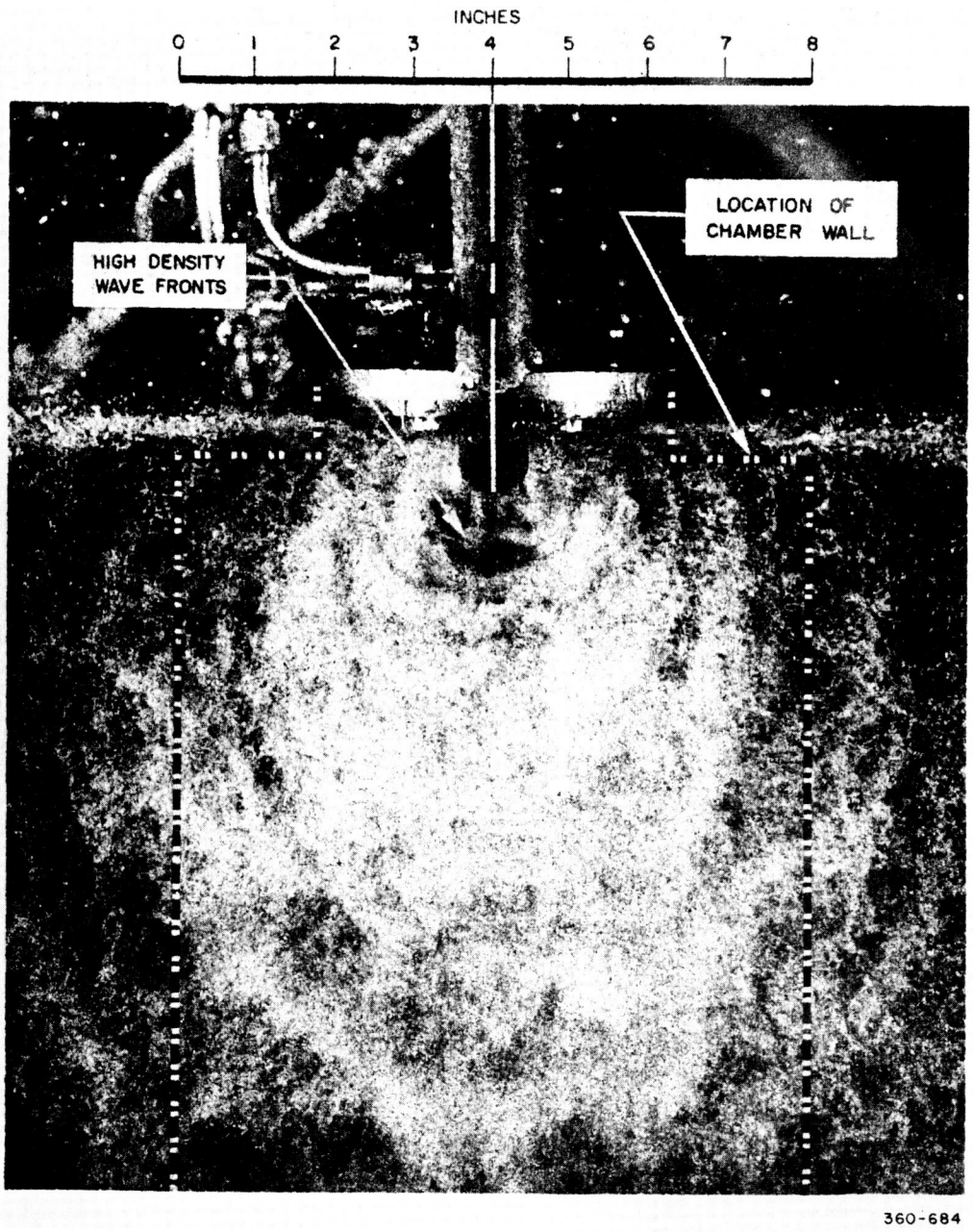
Large-Thrust Transparent-Chamber Test-Installation Schematic

Figure 24



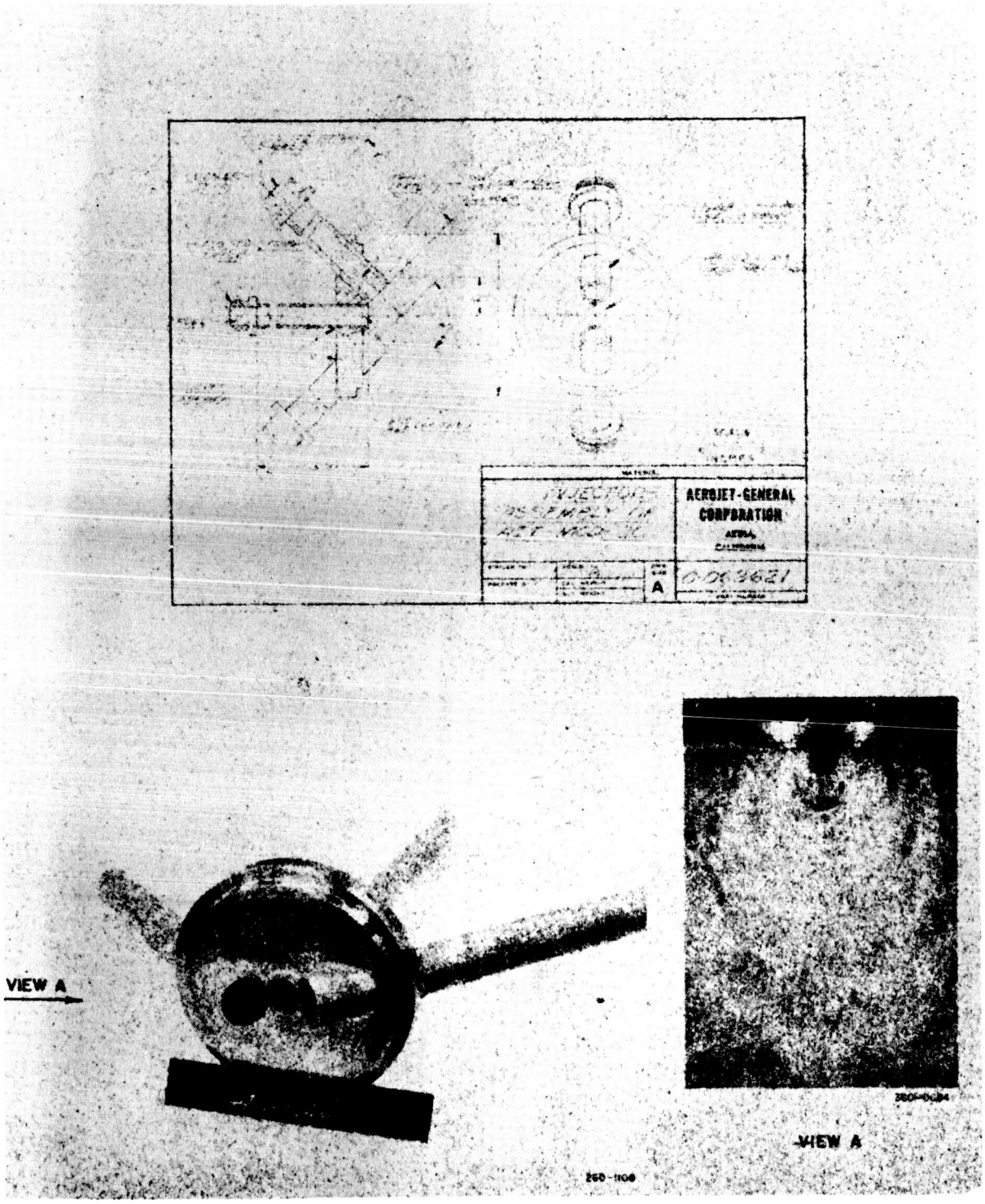
Transparent Thrust Chamber

Figure 25



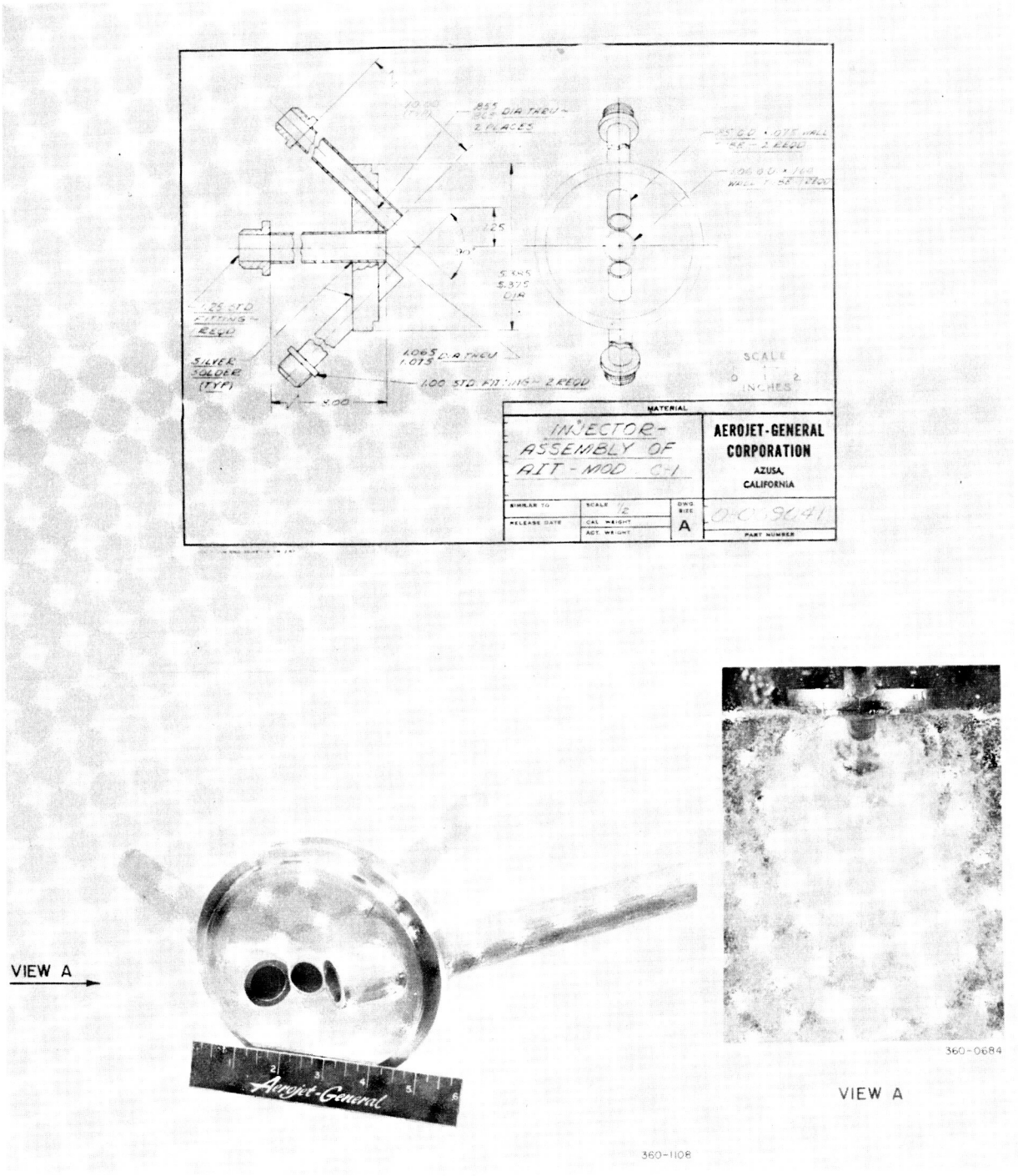
Triplet Spray Pattern AIT Mod 2C

Figure 26



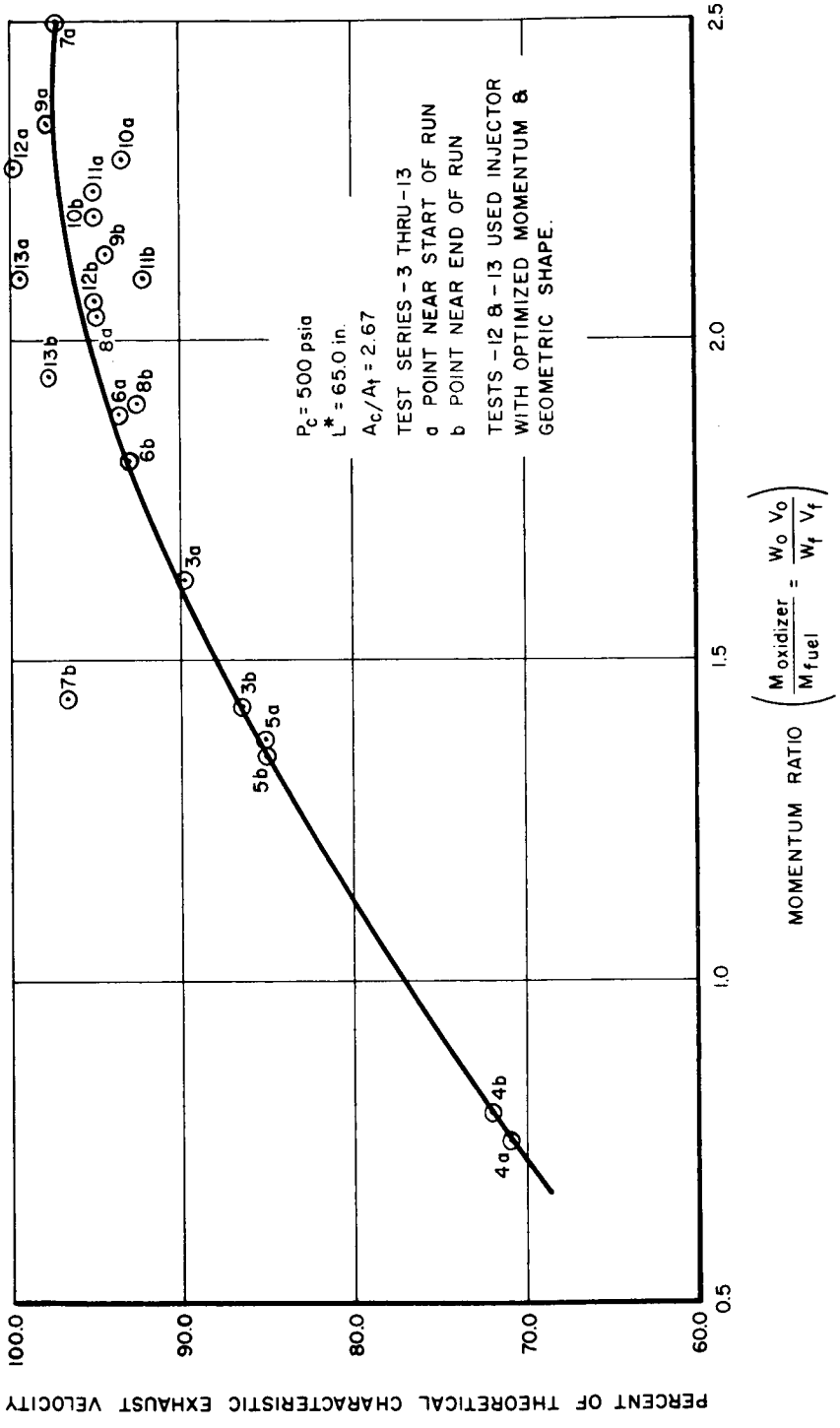
Design and Spray Characteristics AIT Mod 2C Injector Element

Figure 27



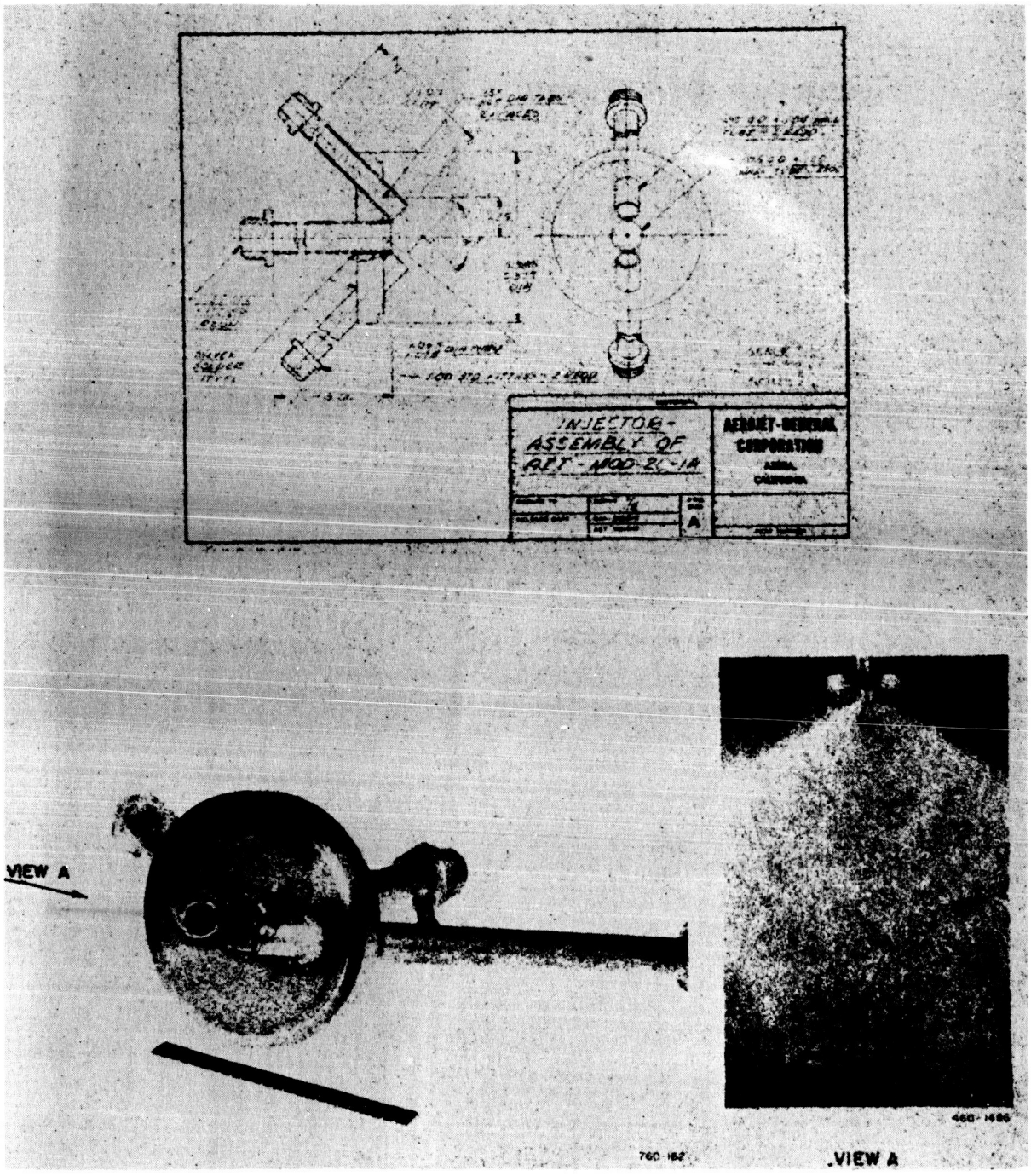
Design and Spray Characteristics AIT Mod 2C-1 Injector Element

Figure 28



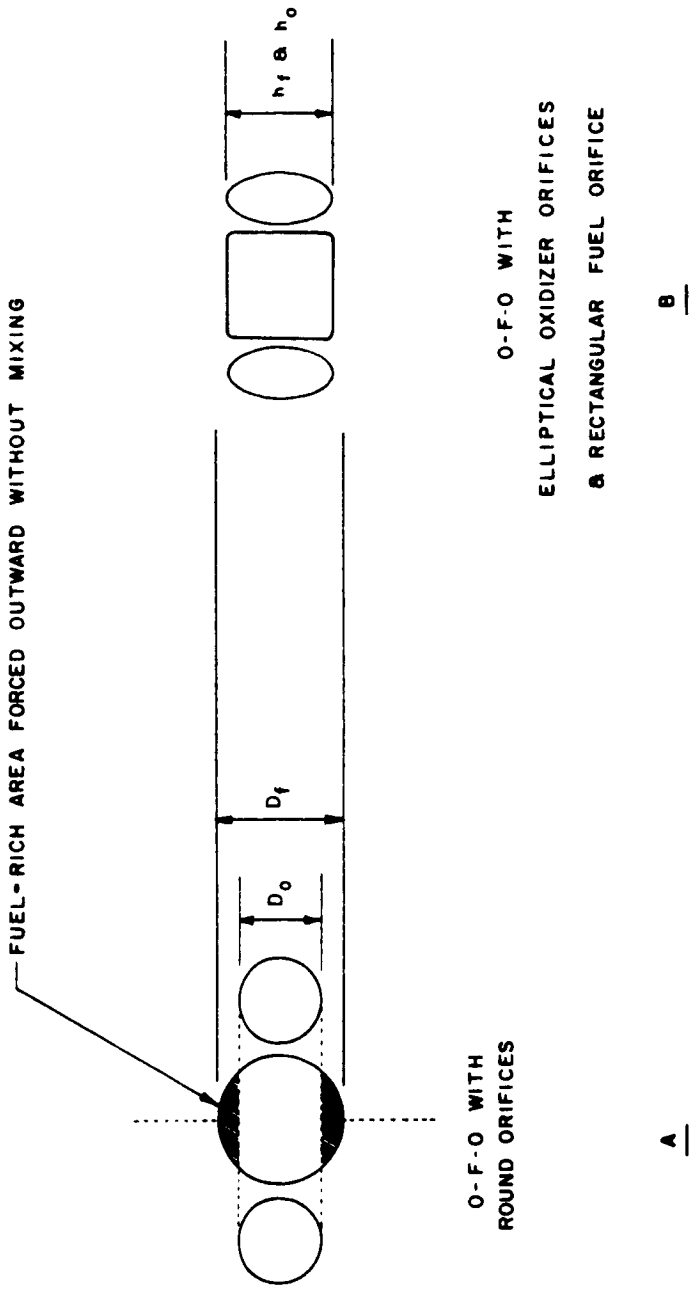
Percent of Theoretical Characteristic Exhaust Velocity vs Momentum Ratio for N_2O_4/Aerozine-50

Figure 29



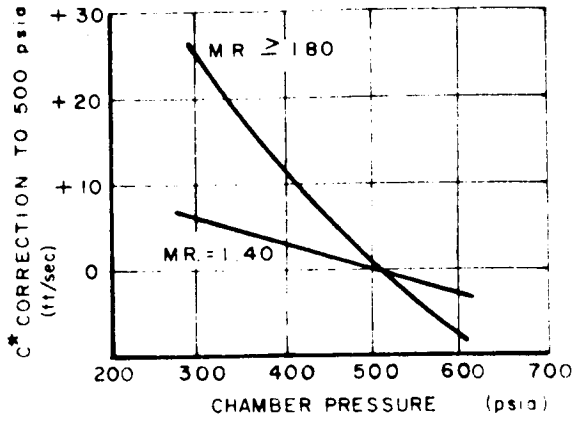
Design and Spray Characteristics AIT Mod 2C-1A Injector Element

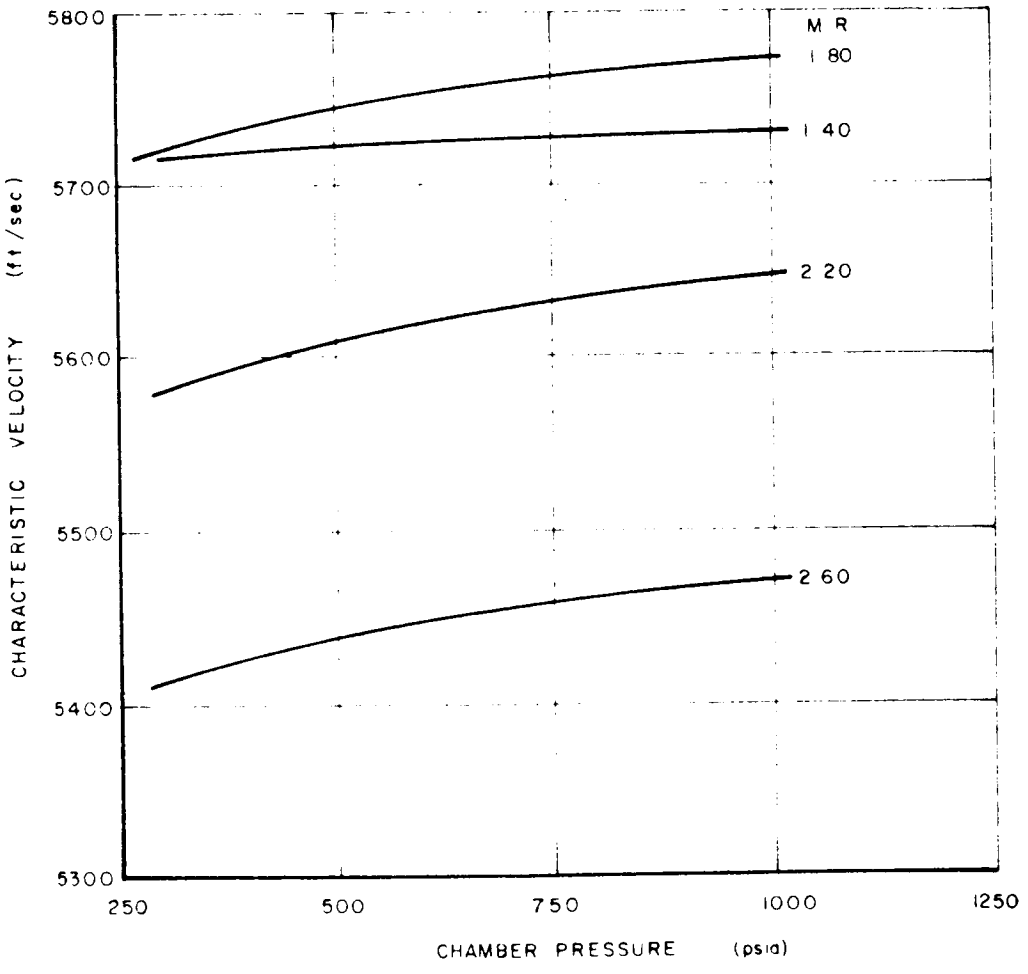
Figure 30



Comparison of Triplet Elements

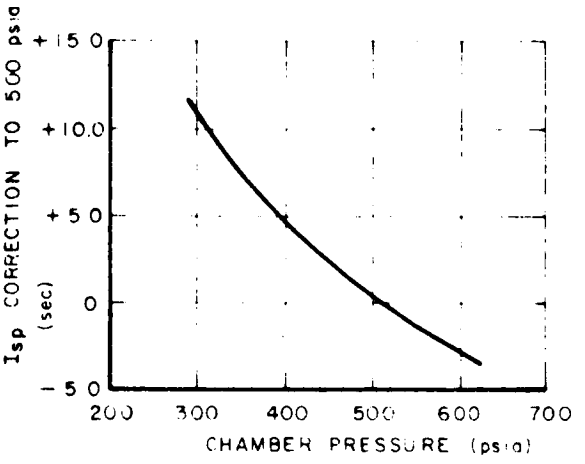
Figure 31

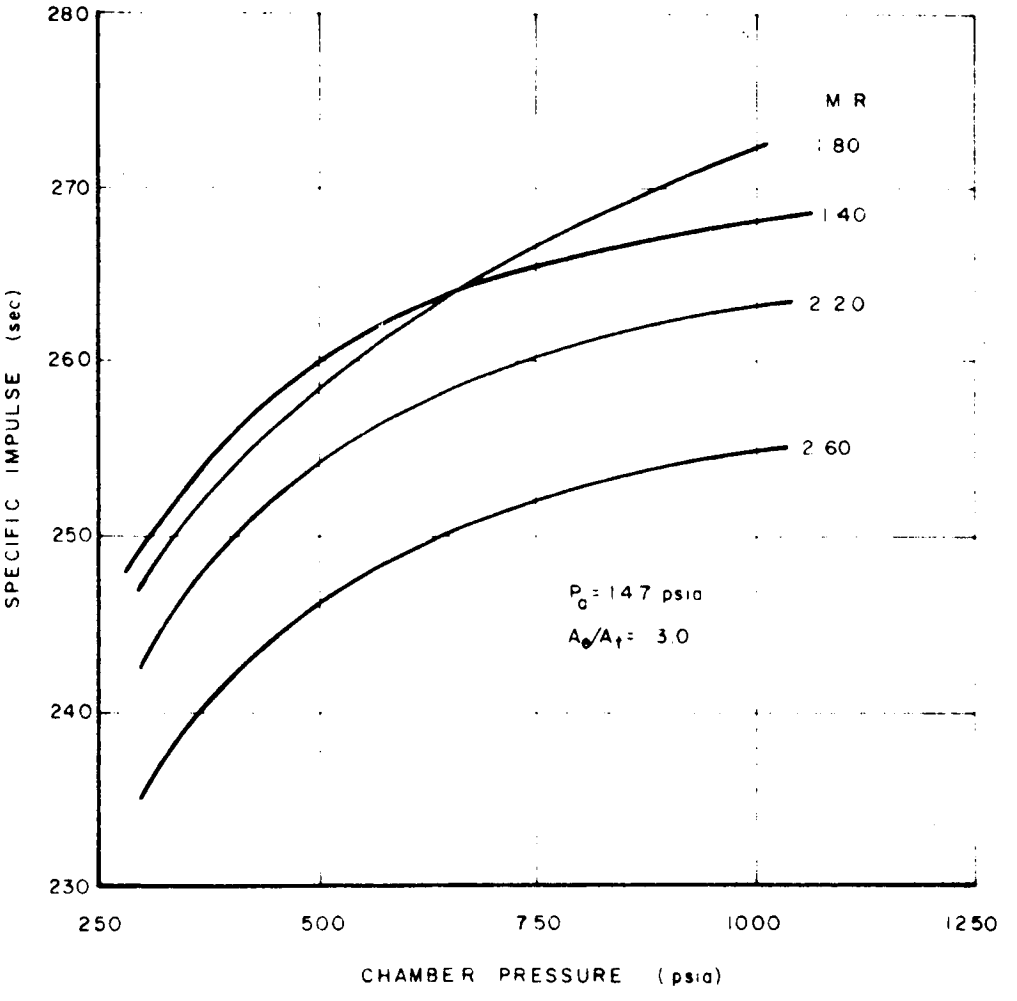




Theoretical Characteristic Velocity vs P_c for N_2O_4/Aerozine-50

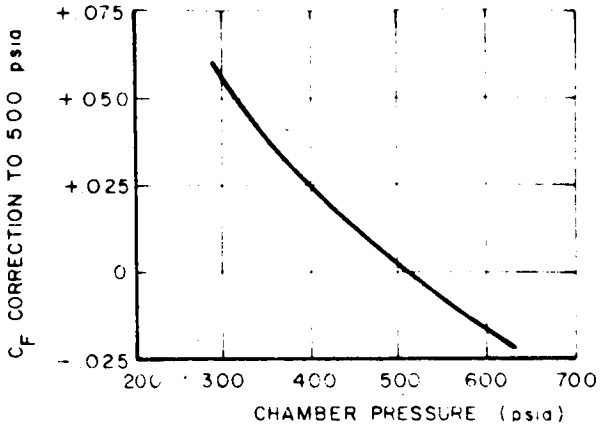
Figure 32

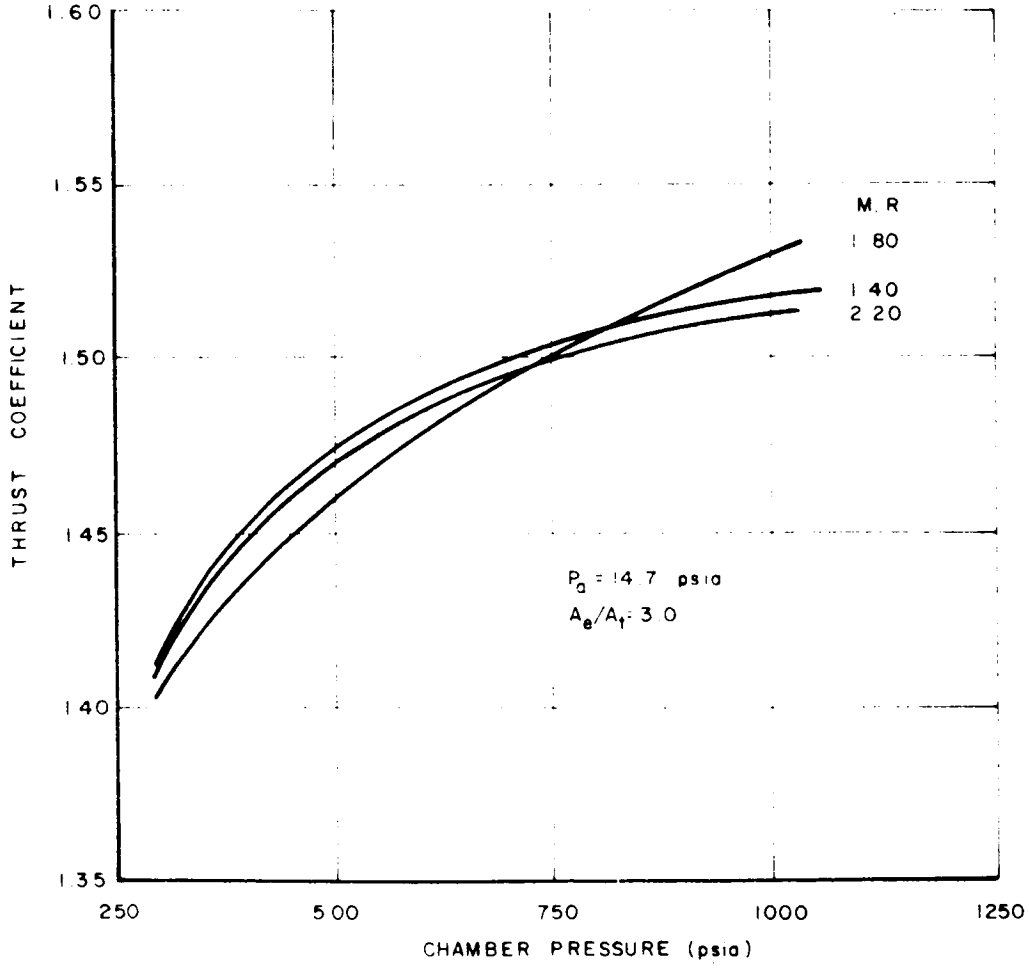




Specific Impulse vs Chamber Pressure for N_2O_4/Aerozine-50

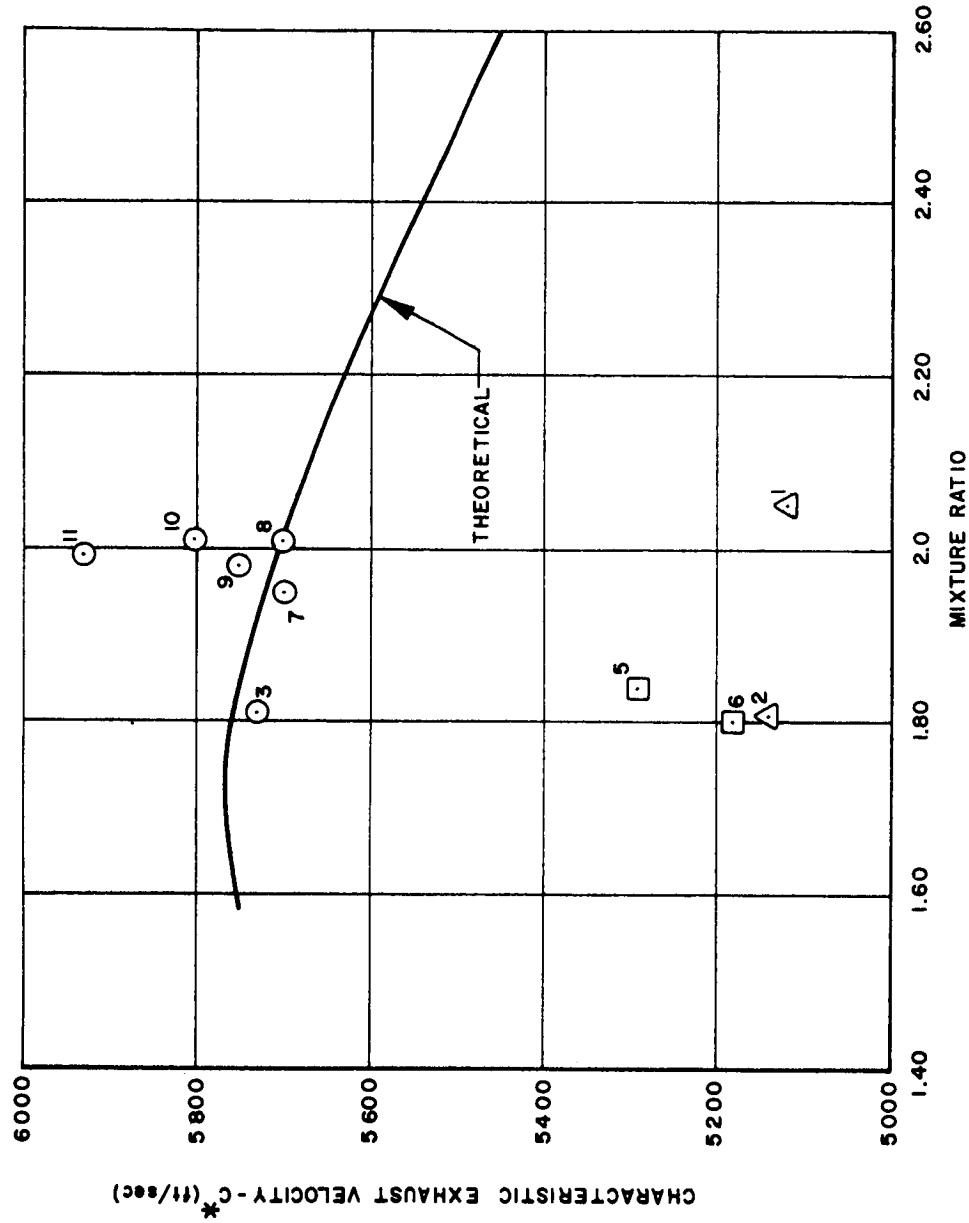
Figure 33





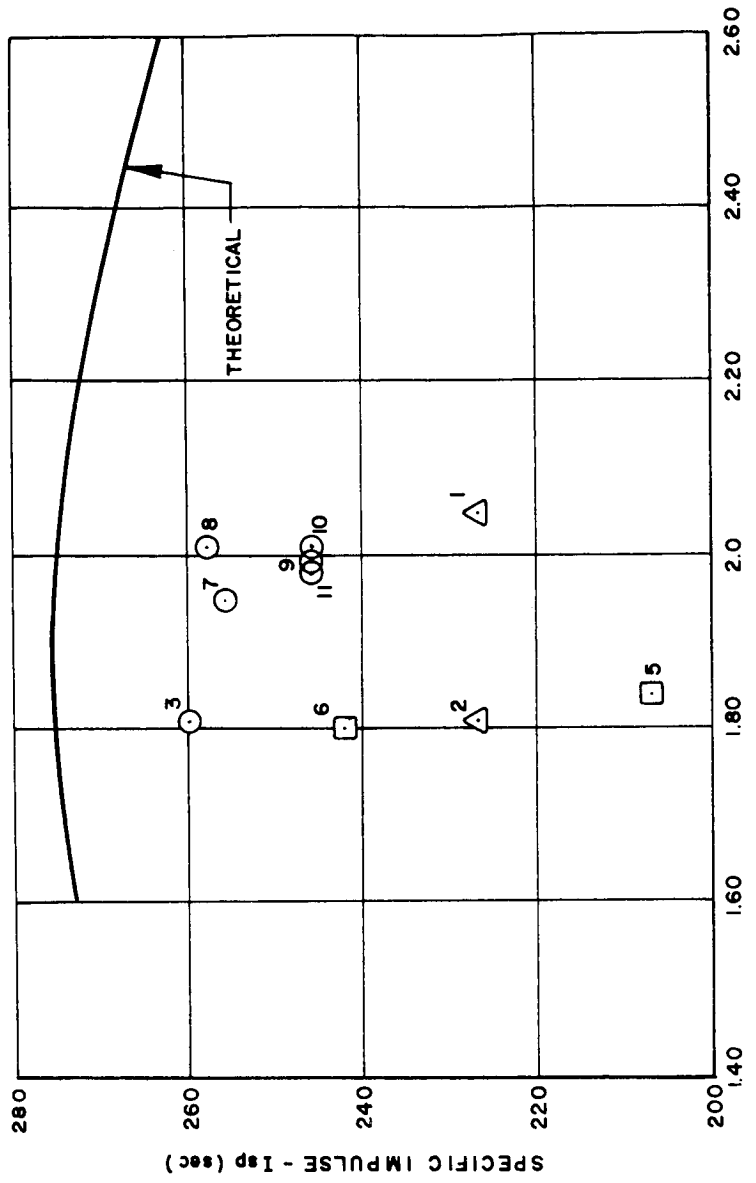
Thrust Coefficient vs Chamber Pressure for N_2O_4/Aerozine-50

Figure 34



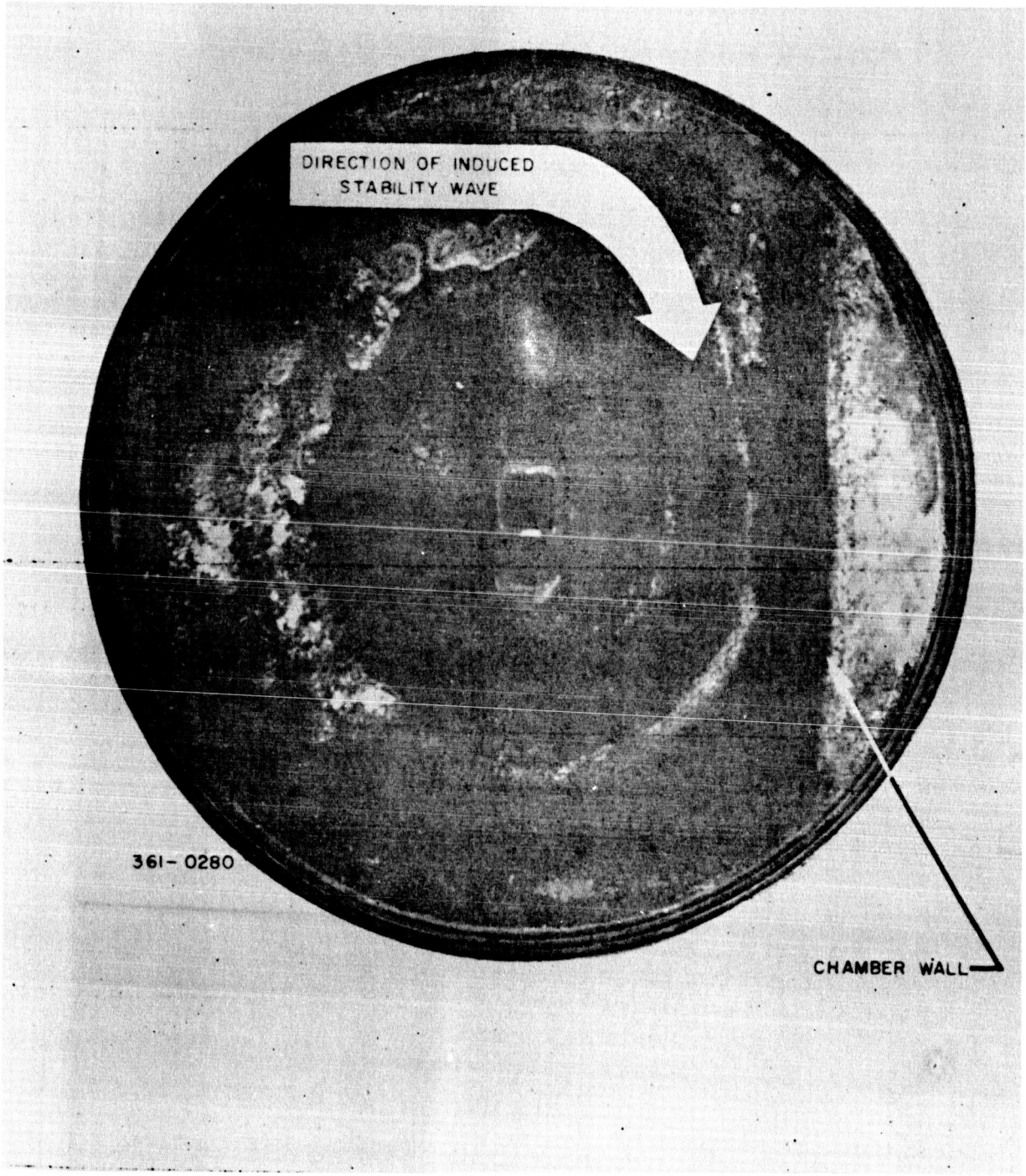
c* vs Mixture Ratio for N_2O_4/Aerozine-50 Large Thrust per Element Injector Program

Figure 35



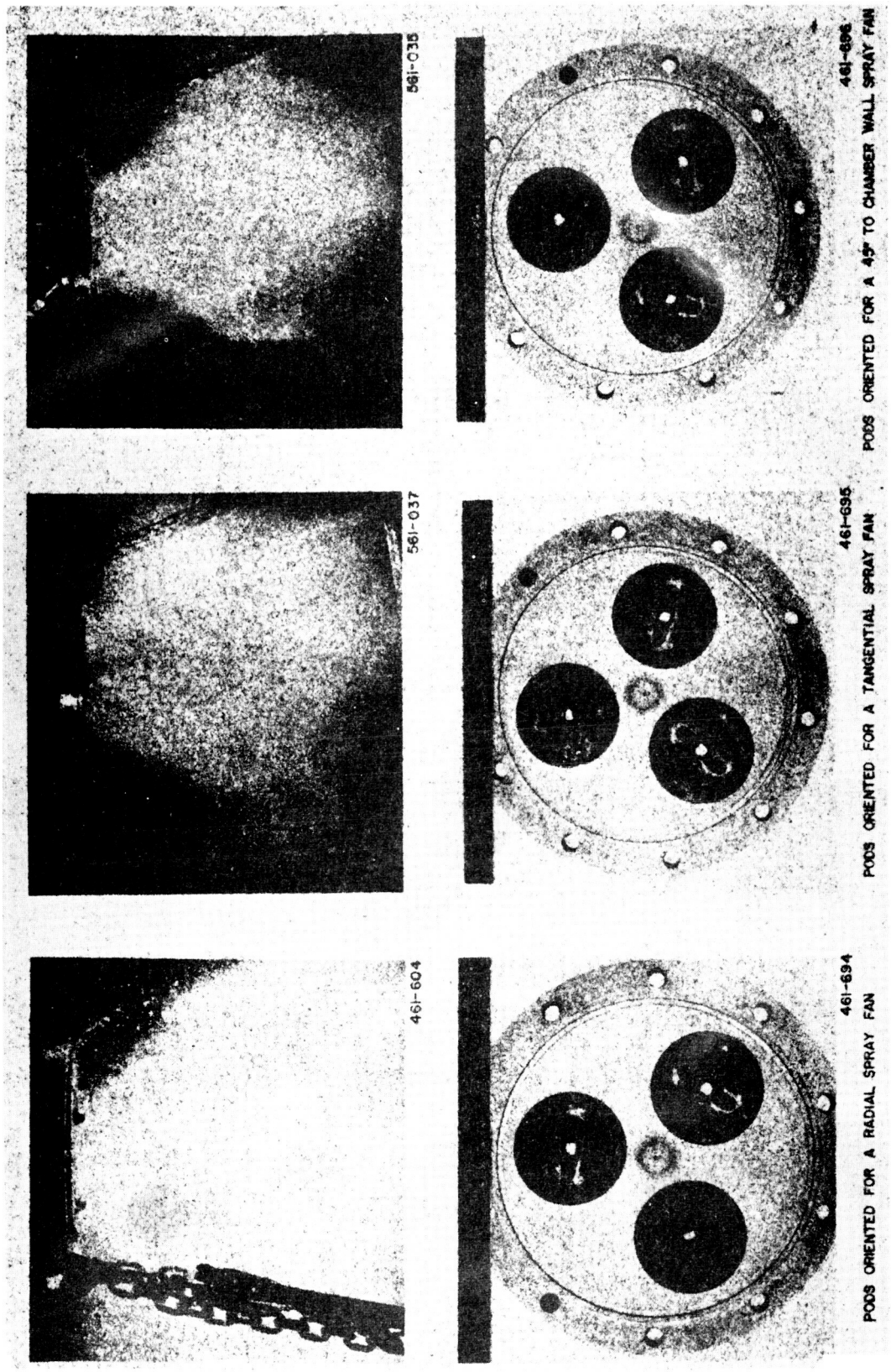
I_{sp} vs Mixture Ratio for N_2O_4/Aerozine-50 Large Thrust per Element Injector Program

Figure 36



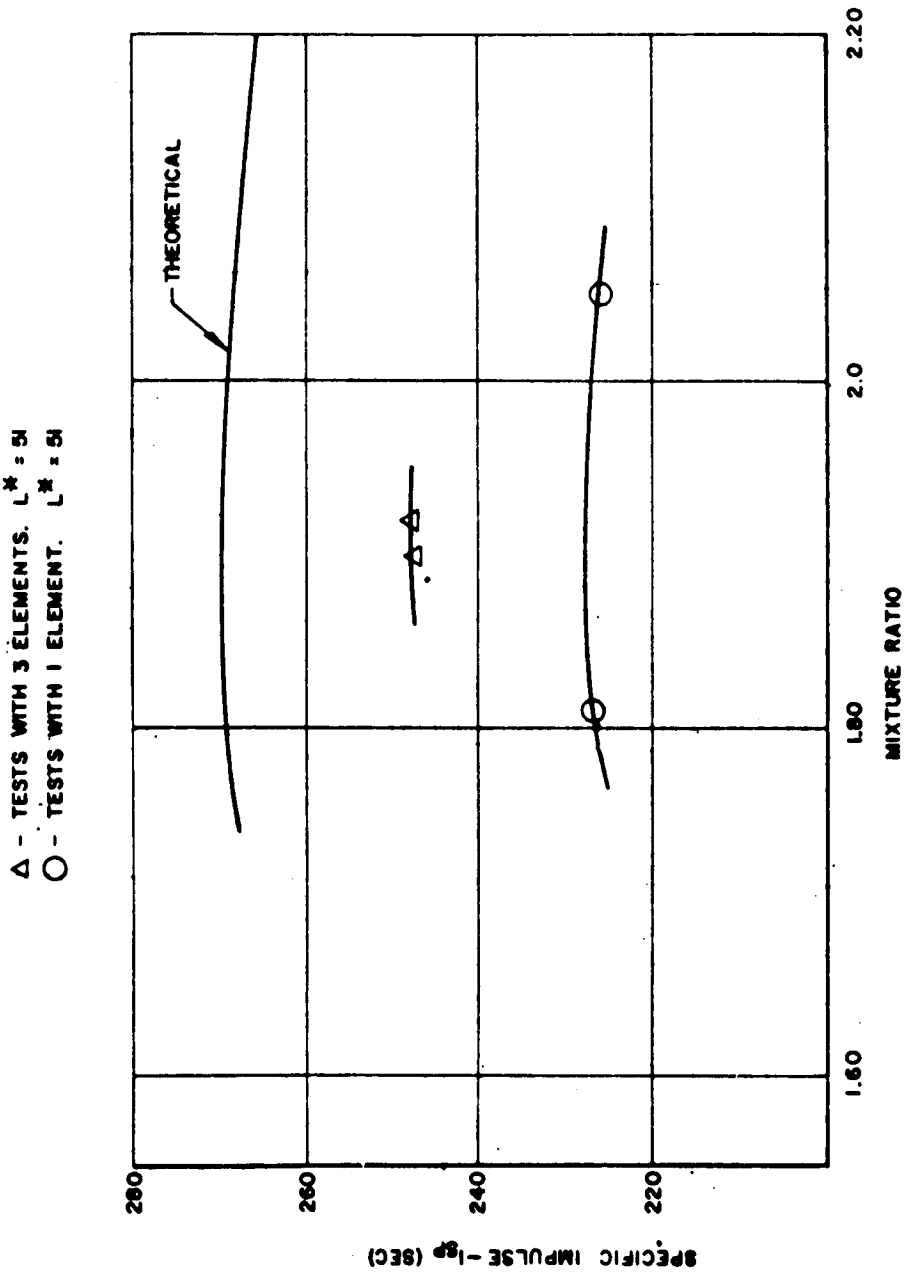
Eroded Injector Element

Figure 37



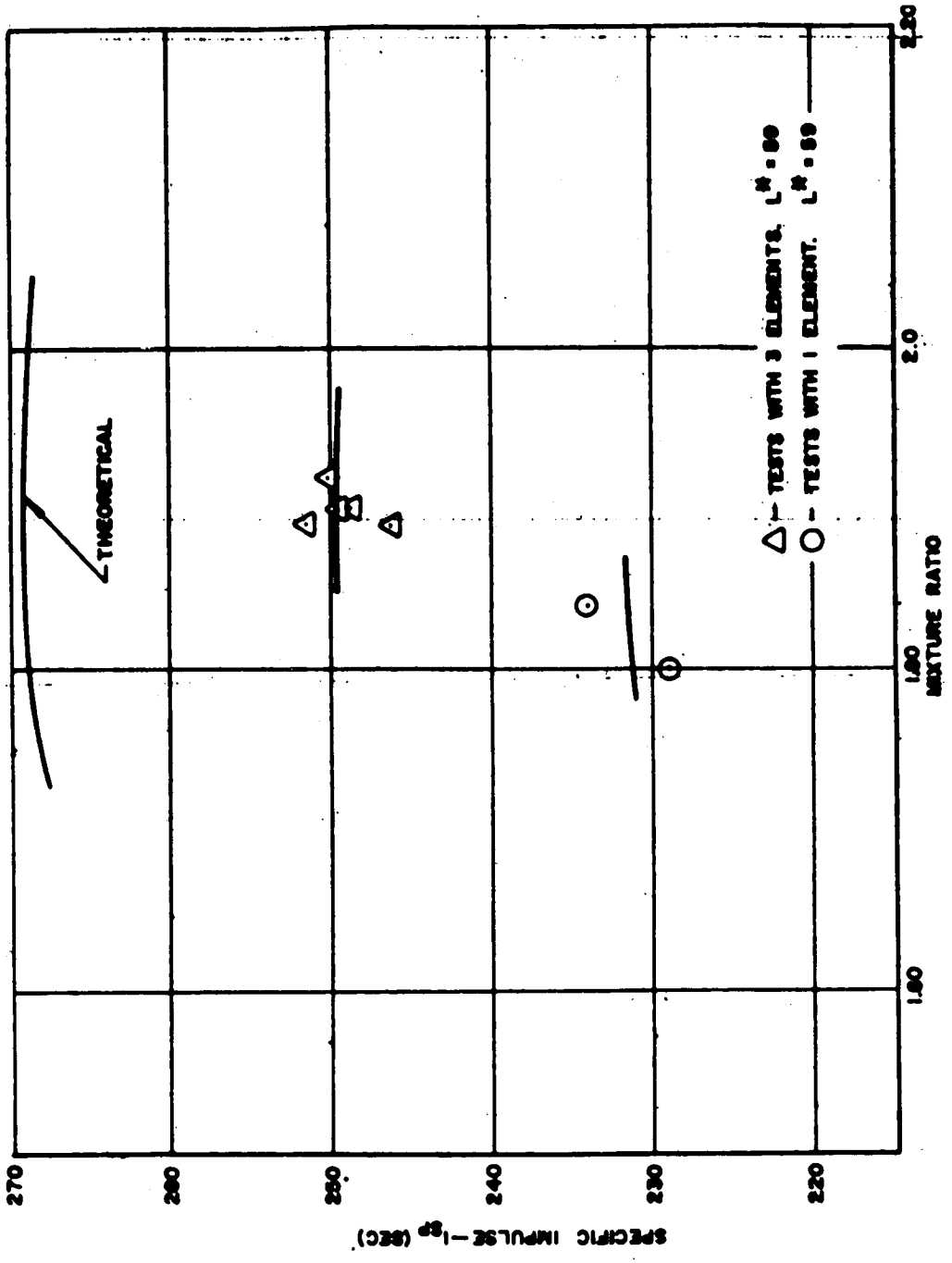
Spray Patterns of 3 POD LT/E Injector

Figure 38



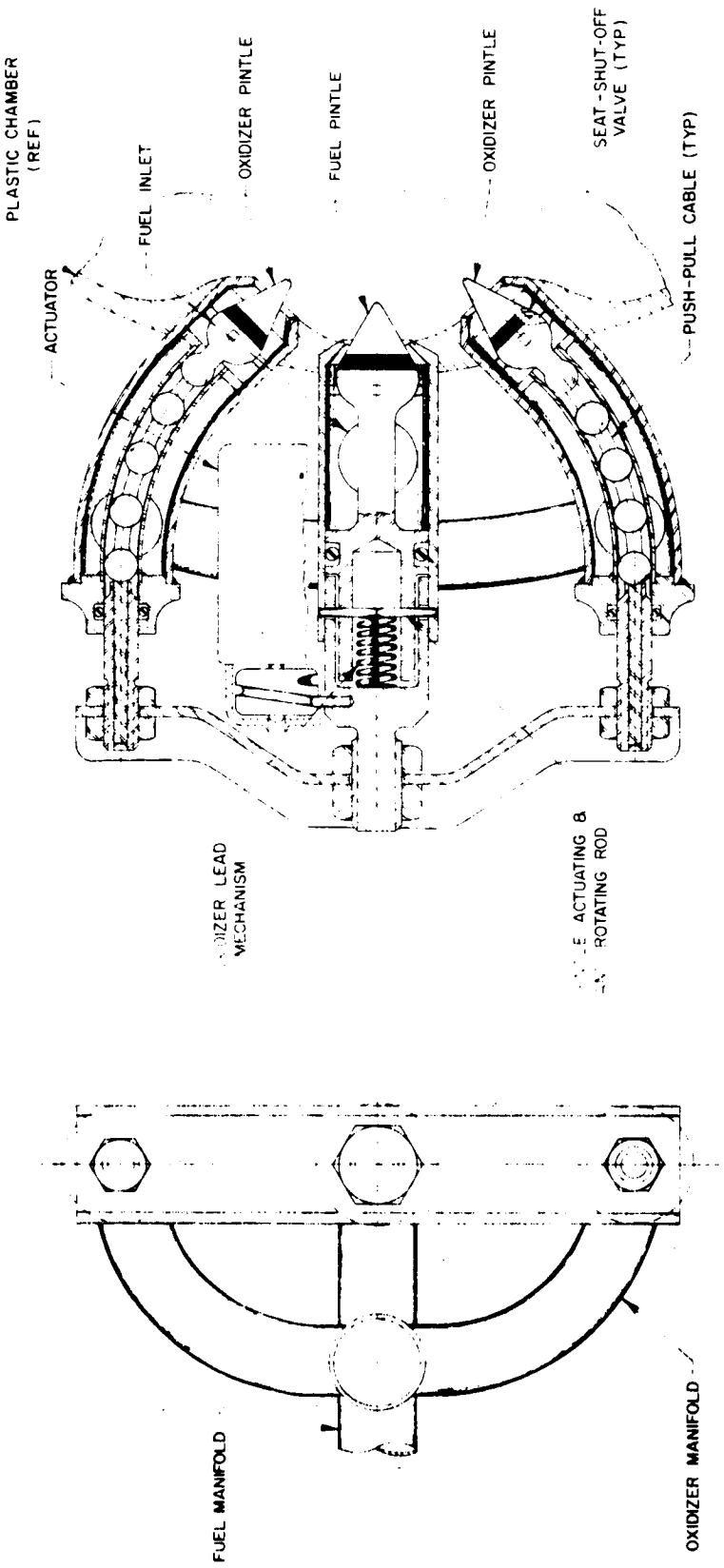
Comparison of 1-Element Injector to 3-Element Injector Performance (L* = 51)

Figure 39



Comparison of 1-Element Injector to 3-Element Injector Performance (L* = 59)

Figure 40



General Configuration of Triplet Injector Assembly (Push-Pull Cable)

Figure 41

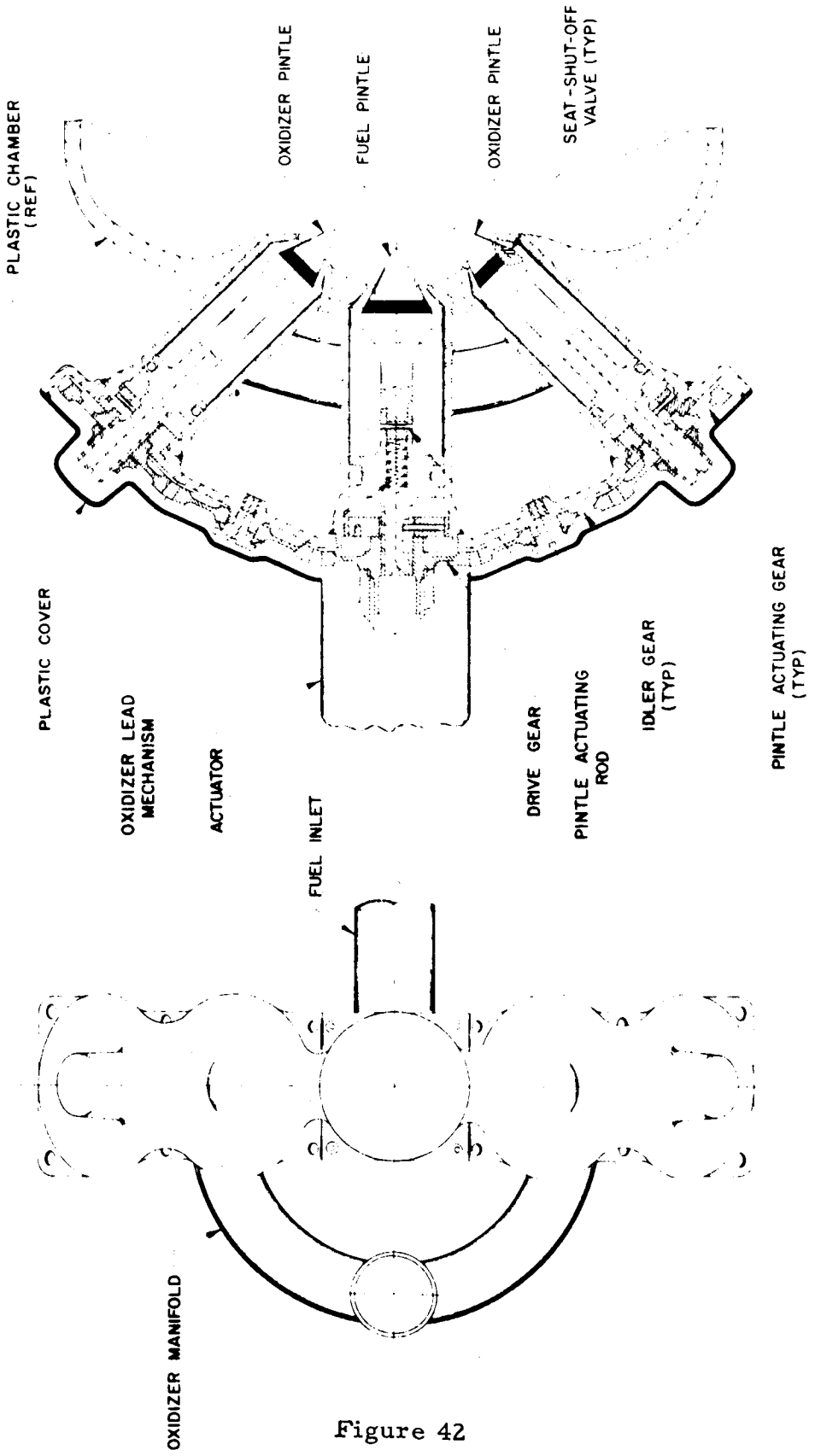
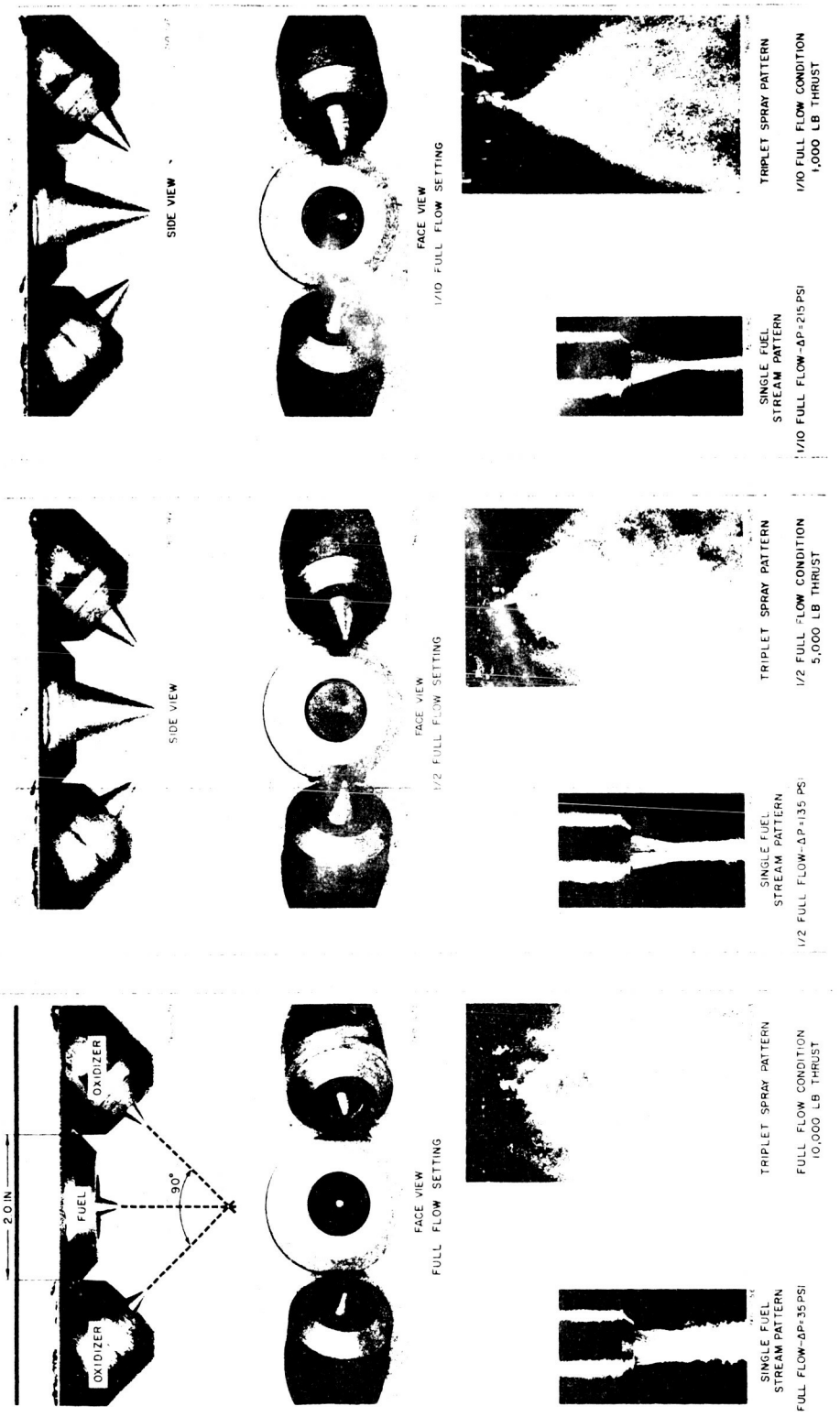


Figure 42

General Configuration of Triplet Injector Assembly (Gear Train)



Water Spray Characteristics of 1,000 to 10,000 lb Thrust Variable Flow Injector

Figure 43

APPENDIX E

RELIABILITY COMPARISONS

TABLE OF CONTENTS

		PAGE NO.
I.	PUMP-FED ENGINES	I-1
II.	PUMP-FED COMPARED TO PRESSURE-FED PROPULSION SYSTEMS	II-1

FIGURE LIST

	FIGURE NO.
XLR87-AJ-3 Rocket Engine	1
HS-10 Rocket Engine	2

I. PUMP-FED ENGINES

A. The objective of this analysis is to compare the potential end product reliability of four rocket engine designs, designated as the YLR87-AJ-3, the HS-10, and the 3200K-8.

These systems were selected as being typical for the range of engine types being considered. Results permit evaluations to be made between the following as to their relative effects upon reliability:

1. Conventional and advanced design
2. Multiple and single combustion chambers
3. Multiple and single turbopumps
4. Suction line and discharge line valving

The YLR87-AJ-3 engine delivers 3,200,000 lb of thrust* (Figure 1). It is a scale-up of the existing Titan first-stage engine, modified for use with LOX/LH_2. This engine reflects current technology in every respect and consists of two 1,600,000-lb thrust engines that are independent of each other, except for a common propellant utilization system and frame.

The HS-10 engine (Figure 2) is a single combustion chamber, single pumping system design identical, except for thrust, with the standard engine of this study.

* This analysis was completed for designs at the 3,200,000-lb thrust level. It was revised to account for the effect of thrust level, and is considered fully applicable for a thrust range from one-million to eight-million pounds.

I, Pump-Fed Engines (cont.)

The 3200K-4 is a heated hydrogen cycle engine using a single pumping system, suction valving, four radially arranged scoop-nozzle thrust chambers, and a radiation-cooled nozzle skirt. Thrust vector control is accomplished by secondary injection.

The 3200K-8 engine is identical with the 3200K-4, except that eight combustors are used instead of four.

B. To make the comparison, the YLR87-AJ-3 engine was selected as a standard because of the vast experience gained during its development, also, its characteristics are the best known. The proposed designs for the HS-10, the 3200K-4 (four-combustor), and the 3200K-8 (eight-combustor) engines were individually compared with this standard. Each design was divided into six major subsystems:

1. Control system
2. Thrust chamber assembly
3. Turbopump assembly
4. Turbopump gas drive system
5. Propellant transfer system
6. Frame

This subsystem breakdown is different from that in common use because all valves, including thrust chamber and gas generator valves, are included in the control system. Furthermore, in both 3200K designs, the heat exchangers in the combustors are used to heat the hydrogen which drives the turbine; therefore, they were classified as gas generators.

I, Pump-Fed Engines (cont.)

C. Each subsystem of the standard design was compared with the corresponding subsystem in each of the other three designs. This comparison was based upon 13 attributes, each of which contributes in some degree to the over-all complexity (hence, to the failure potential) of the subsystem:

1. Functional operations
2. External environment
3. Internal environment
4. Moving parts
5. Nonmoving parts
6. Material compatibility
7. Safety margins
8. Dimensional tolerances
9. Redundancy
10. Current technology
11. Scale factors
12. Modes of failure
13. Ease of fabrication

Each attribute was assigned a weight (on the basis of a 0.0 to 1.0 scale) which specifies its relative importance in contributing to the over-all subsystem complexity.

I, Pump-Fed Engines (cont.)

Individual attributes for the standard design were arbitrarily assigned a comparison number of 100. A comparison number was then assigned to each of the other systems to reflect the relative degree that the attribute in question contributes to the complexity of the subsystem under study. For example, if the turbopump of one system has ten moving parts while that of a second system has only five moving parts, the comparison numbers assigned to this attribute for the turbopumps of these two systems would be 100 and 50, respectively. The comparison factor for each attribute of the subsystem under study is the product of the corresponding attribute weight and comparison number. The subsystem complexity number is the sum of the corresponding comparison factors for all of the attributes.

Each subsystem was assigned a subsystem weight (on the basis of a 0.0 to 1.0 scale) which specifies the relative degree to which the subsystem contributes to system failure. The complexity factor of the subsystem under study is the product of the subsystem weight and the complexity number, and the system complexity factor is the sum of the corresponding subsystem complexity factors. Ratio of system failure rates is the same as the ratio of the corresponding system complexity factors, and the subsystem failure rates have been apportioned from the system failure rates by appropriate techniques.

Detailed tabulations of these figures appear in subsequent pages.

RELIABILITY COMPARISON

	Subsystem	FAILURES/100,000* 87-3	HS-10	3200K-4	3200K-8
1.	Controls	1,783	1,631	1,713	1,773
2.	TCA	1,251	1,658	1,297	1,491
3.	TPA	1,251	852	710	712
4.	GGA	416	390	331	343
5.	Propellant transfer	386	355	313	314
6.	Frame	8	8	8	8
	System Failures	5,000	4,807	4,304	4,564

* Relative values based on the assigned reliability goal of 0.95 for the YLR87-AJ-3 engine.

RELIABILITY COMPARISON

	Subsystem	Weight	COMPLEXITY FACTOR* 87-3	HS-10	3200K-4	3200K-8
1.	Controls	1.0	920	842	887	917
2.	TCA	0.7	644	856	670	770
3.	TPA	0.7	644	438	366	366
4.	Gas generator	0.3	213	200	170	176
5.	Propellant transfer	0.3	198	182	161	161
6.	Frame	0.01	4	4	4	4
	TOTAL		2,623	2,522	2,258	2,394

* Complexity factor equals complexity number times the subsystem weight

RELIABILITY COMPARISON

Subsystem Controls

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	1.0	100	90	100	100	100	90	100	100
Environment, external	0.3	100	80	70	70	30	24	21	21
Environment, internal	1.0	100	110	100	100	100	110	100	100
Moving parts	1.0	100	60	100	100	100	60	100	100
Nonmoving parts	0.6	100	50	50	100	60	30	30	60
Material compatibility	1.0	100	100	100	100	100	100	100	100
Safety margin	0.1	100	100	100	100	10	10	10	10
Dimensional tolerances	1.0	100	100	120	120	100	100	120	120
Redundancy	1.0	100	100	90	90	100	100	90	90
State-of-the-art	0.8	100	110	120	120	80	88	96	96
Scale factors	0.1	100	100	100	100	10	10	10	10
Modes of failure	1.0	100	90	80	80	100	90	80	80
Ease of fabrication	0.3	100	100	100	100	30	30	30	30
TOTAL (subsystem complexity number)						920	842	887	917

* Attribute weight x comparison number.

CONFIDENTIAL

RELIABILITY COMPARISON

Subsystem Thrust Chamber Assembly

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	1.0	100	120	110	110	100	120	110	110
Environment, external	0.1	100	120	120	120	10	12	12	12
Environment, internal	1.0	100	100	100	100	100	100	100	100
Moving parts	0.1	100	50	120	120	10	5	12	12
Nonmoving parts	0.6	100	50	150	400	60	30	90	240
Material compatibility	0.8	100	100	100	100	80	80	80	80
Safety margin	0.9	100	60	40	40	90	54	36	36
Dimensional tolerances	0.8	100	120	50	50	80	96	40	40
Redundancy	0.0	-	-	-	-	-	-	-	-
State-of-the-art	0.9	100	130	130	130	90	117	117	117
Scale factors	1.0	100	200	120	120	100	200	120	120
Modes of failure	1.0	100	130	110	110	100	130	110	110
Ease of fabrication	1.0	100	180	140	140	100	180	140	140
TOTAL (subsystem complexity number)						920	1,224	957	1,100

* Attribute weight x comparison number.

CONFIDENTIAL

RELIABILITY COMPARISON

Subsystem Turbopump Assembly

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	1.0	100	90	80	80	100	90	80	80
Environment, external	0.1	100	100	100	100	10	10	10	10
Environment, internal	1.0	100	120	80	80	100	120	80	80
Moving parts	1.0	100	20	25	25	100	20	25	25
Nonmoving parts	1.0	100	40	40	40	100	40	40	40
Material compatibility	0.4	100	50	50	50	40	20	20	20
Safety margin	0.5	100	60	40	40	50	30	20	20
Dimensional tolerances	1.0	100	70	30	30	100	70	30	30
Redundancy	0.0	-	-	-	-	-	-	-	-
State-of-the-art	0.8	100	100	110	110	80	80	88	88
Scale factors	0.9	100	100	100	100	90	90	90	90
Modes of failure	1.0	100	30	20	20	100	30	20	20
Ease of fabrication	0.5	100	50	40	40	50	25	20	20
TOTAL (subsystem complexity number)						920	625	523	523

* Attribute weight x comparison number

CONFIDENTIAL

RELIABILITY COMPARISON

Subsystem Gas Generating System

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	1.0	100	100	70	70	100	100	70	70
Environment, external	0.1	100	100	70	70	10	10	7	7
Environment, internal	1.0	100	100	150	150	100	100	150	150
Moving parts	0.0	-	-	-	-	-	-	-	-
Nonmoving parts	0.3	100	50	100	200	30	15	30	60
Material compatibility	0.4	100	100	100	100	40	40	40	40
Safety margin	0.3	100	60	40	40	30	18	12	12
Dimensional tolerances	0.1	100	100	120	120	10	10	12	12
Redundancy	1.0	100	100	20	10	100	100	20	10
State-of-the-art	0.4	100	100	200	200	40	40	80	80
Scale factors	0.7	100	100	100	100	70	70	70	70
Modes of failure	1.0	100	100	50	50	100	100	50	50
Ease of fabrication	0.8	100	80	30	30	80	64	24	24
TOTAL (subsystem complexity number)						710	667	565	585

* Attribute weight x comparison number

CONFIDENTIAL

RELIABILITY COMPARISON

Subsystem Propellant Transfer

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	0.6	100	100	70	70	60	60	42	42
Environment, external	0.1	100	100	100	100	10	10	10	10
Environment, internal	1.0	100	120	110	110	100	120	110	110
Moving parts	0.0	-	-	-	-	-	-	-	-
Nonmoving parts	1.0	100	60	50	60	100	60	50	60
Material compatibility	0.9	100	100	100	100	90	90	90	90
Safety margin	0.8	100	100	70	70	80	80	56	56
Dimensional tolerances	0.9	100	60	100	100	90	54	90	90
Redundancy	0.0	-	-	-	-	-	-	-	-
State-of-the-art	0.1	100	130	80	80	10	13	8	8
Scale factors	0.1	100	100	100	100	10	10	10	10
Modes of failure	1.0	100	100	70	70	100	100	70	70
Ease of fabrication	0.1	100	110	110	110	10	11	11	11
TOTAL (subsystem complexity number)						660	608	537	547

* Attribute weight x comparison number

RELIABILITY COMPARISON

Subsystem Frame

ATTRIBUTES		COMPARISON NO.				COMPARISON FACTOR*			
Item	Weight	87-3	HS-10	3200K-4	3200K-8	87-3	HS-10	3200K-4	3200K-8
Functional operations	0.7	100	100	100	100	70	70	70	70
Environment, external	0.6	100	100	100	100	60	60	60	60
Environment, internal	0.0	-	-	-	-	-	-	-	-
Moving parts	0.0	-	-	-	-	-	-	-	-
Nonmoving parts	0.6	100	100	100	100	60	60	60	60
Material compatibility	0.0	-	-	-	-	-	-	-	-
Safety margin	1.0	100	100	100	100	100	100	100	100
Dimensional tolerances	0.1	100	100	100	100	10	10	10	10
Redundancy	0.0	-	-	-	-	-	-	-	-
State-of-the-art	0.2	100	100	100	100	20	20	20	20
Scale factors	1.0	100	100	100	100	100	100	100	100
Modes of failure	0.1	100	100	100	100	10	10	10	10
Ease of fabrication	0.1	100	100	100	100	10	10	10	10
TOTAL (subsystem complexity number)						440	440	440	440

* Attribute weight x comparison number

CONFIDENTIAL

II. PUMP-FED COMPARED TO PRESSURE-FED PROPULSION SYSTEMS

The objective of this analysis was to compare potential end product reliability of a pressure-fed 6,700,000-lb engine with a pump-fed 3,200,000-lb engine with allowances made for an increase in thrust to 6,700,000-lb.

To carry out the comparison, the eight combustor version of the 3,200,000-lb engine was selected as the standard. Upgrading of thrust to 6,700,000-lb was accomplished by increasing the thrust in each combustor with the number of combustors remaining unchanged. The proposed design of the pressure-fed 6,700,000-lb engine was then compared with this upgraded 3,200,000-lb standard engine.

The same comparison analysis as described in Part I is used, except that the 3,200,000-lb engine is the standard. Detailed tabulation of the comparison, complexity and failure probability figures appear in subsequent pages. A consolidated parts list follows.

RELIABILITY COMPARISON

	Subsystem	Failures/100,000* 3200K	6700K
1.	Controls	1,695	1,853
2.	Propellant feed	1,358	997
3.	Thrust chamber assembly	1,190	1,238
4.	Propellant transfer	512	602
5.	Tankage, structure, and frame	341	384
	System Failures	5,000	4,979

* Relative values based on an assumed reliability of 0.95 for the 3200K standard engine.

RELIABILITY COMPARISON

	Subsystem	Subsystem Weight	Complexity Factors 3200K*	Complexity Factors 6700K**
1.	Controls	1.0	870	952
2.	Propellant feed	0.8	696	510
3.	Thrust chamber assembly	0.7	609	634
4.	Propellant transfer	0.3	261	307
5.	Tankage, structure, and frame	0.2	174	196
TOTALS (system complexity factors)			2,610	2,599

* The ratios of these subsystem complexity factors are the same as the corresponding ratios of subsystem weights.

** The ratio of the 6700K complexity factor for one subsystem to that for the 3200K engine is the same as the corresponding ratio of complexity numbers.

RELIABILITY COMPARISON

Subsystem Controls

ATTRIBUTES		COMPARISON NUMBER		COMPARISON FACTOR*	
Item	Weight	3200K	6700K	3200K	6700K
Functional operations	1.0	100	200	100	200
Environment, external	0.2	100	100	20	20
Environment, internal	1.0	100	100	100	100
Moving parts	0.8	100	70	80	56
Nonmoving parts (including seals)	1.0	100	100	100	100
Material compatibility	0.6	100	110	60	66
Safety margin	0.2	100	100	20	20
Dimensional tolerances	1.0	100	100	100	100
State-of-the-art	0.7	100	100	70	70
Scale factors (differences in parameter magnitudes)	0.8	100	100	80	80
Modes of failure	1.0	100	100	100	100
Ease of fabrication	0.4	100	100	40	40
TOTALS (subsystem complexity number)				870	952

* Attribute weight x comparison number

RELIABILITY COMPARISON

Subsystem Thrust Chamber Assembly

ATTRIBUTES		COMPARISON NUMBER		COMPARISON FACTOR*	
Item	Weight	3200K	6700K	3200K	6700K
Functional operations	1.0	100	140	100	140
Environment, external	0.2	100	100	20	20
Environment, internal	1.0	100	60	100	60
Moving parts	0.2	100	90	20	18
Nonmoving parts (including seals)	0.7	100	200	70	140
Material compatibility	0.9	100	100	90	90
Safety margin	0.6	100	80	60	48
Dimensional tolerances	0.5	100	130	50	65
State-of-the-art	0.8	100	130	80	104
Scale factors (differences in parameter magnitudes)	0.5	100	60	50	30
Modes of failure	1.0	100	90	100	90
Ease of fabrication	1.0	100	70	100	70
TOTAL (subsystem complexity number)				840	875

* Attribute weight x comparison number

CONFIDENTIAL

RELIABILITY COMPARISON

Subsystem Propellant Feed System

ATTRIBUTES		COMPARISON NUMBER		COMPARISON FACTOR*	
Item	Weight	3200K	6700K	3200K	6700K
Functional operations	1.0	100	20	100	20
Environment, external	1.0	100	100	100	100
Environment, internal	1.0	100	90	100	90
Moving parts	1.0	100	20	100	20
Nonmoving parts (including seals)	1.0	100	140	100	140
Material compatibility	1.0	100	60	100	60
Safety margin	0.8	100	80	80	64
Dimensional tolerances	0.8	100	70	80	56
State-of-the-art	0.8	100	100	80	80
Scale factors (differences in parameter magnitudes)	0.5	100	120	50	60
Modes of failure	1.0	100	30	100	30
Ease of fabrication	0.3	100	90	30	27
TOTAL (subsystem complexity number)				1,020	747

* Attribute weight x comparison number

RELIABILITY COMPARISON

Subsystem Propellant Transfer System

ATTRIBUTES		COMPARISON NUMBER		COMPARISON FACTOR*	
Item	Weight	3200K	6700K	3200K	6700K
Functional operations	0.0	-	-	-	-
Environment, external	0.1	100	90	10	9
Environment, internal	1.0	100	70	100	70
Moving parts	0.0	-	-	-	-
Nonmoving parts (including seals)	1.0	100	200	100	200
Material compatibility	0.9	100	100	90	90
Safety margin	0.7	100	90	70	63
Dimensional tolerances	1.0	100	110	100	110
State-of-the-art	0.1	100	100	10	10
Scale factors (differences in parameter magnitudes)	0.5	100	100	50	50
Modes of failure	1.0	100	150	100	150
Ease of fabrication	0.3	100	80	30	24
TOTAL (subsystem complexity number)				660	776

* Attribute weight x comparison number

RELIABILITY COMPARISON

Subsystem Tankage, Structure, and Frame

ATTRIBUTES		COMPARISON NUMBER		COMPARISON FACTOR*	
Item	Weight	3200K	6700K	3200K	6700K
Functional operations	0.4	100	120	40	48
Environment, external	0.2	100	100	20	20
Environment, internal	1.0	100	150	100	150
Moving parts	0.0	-	-	-	-
Nonmoving parts (including seals)	1.0	100	70	100	70
Material compatibility	1.0	100	100	100	100
Safety margin	1.0	100	150	100	150
Dimensional tolerances	0.5	100	100	50	50
State-of-the-art	0.0	-	-	-	-
Scale factors (differences in parameter magnitudes)	0.3	100	100	30	30
Modes of failure	1.0	100	100	100	100
Ease of fabrication	0.3	100	120	30	36
TOTALS (subsystem complexity number)				670	754

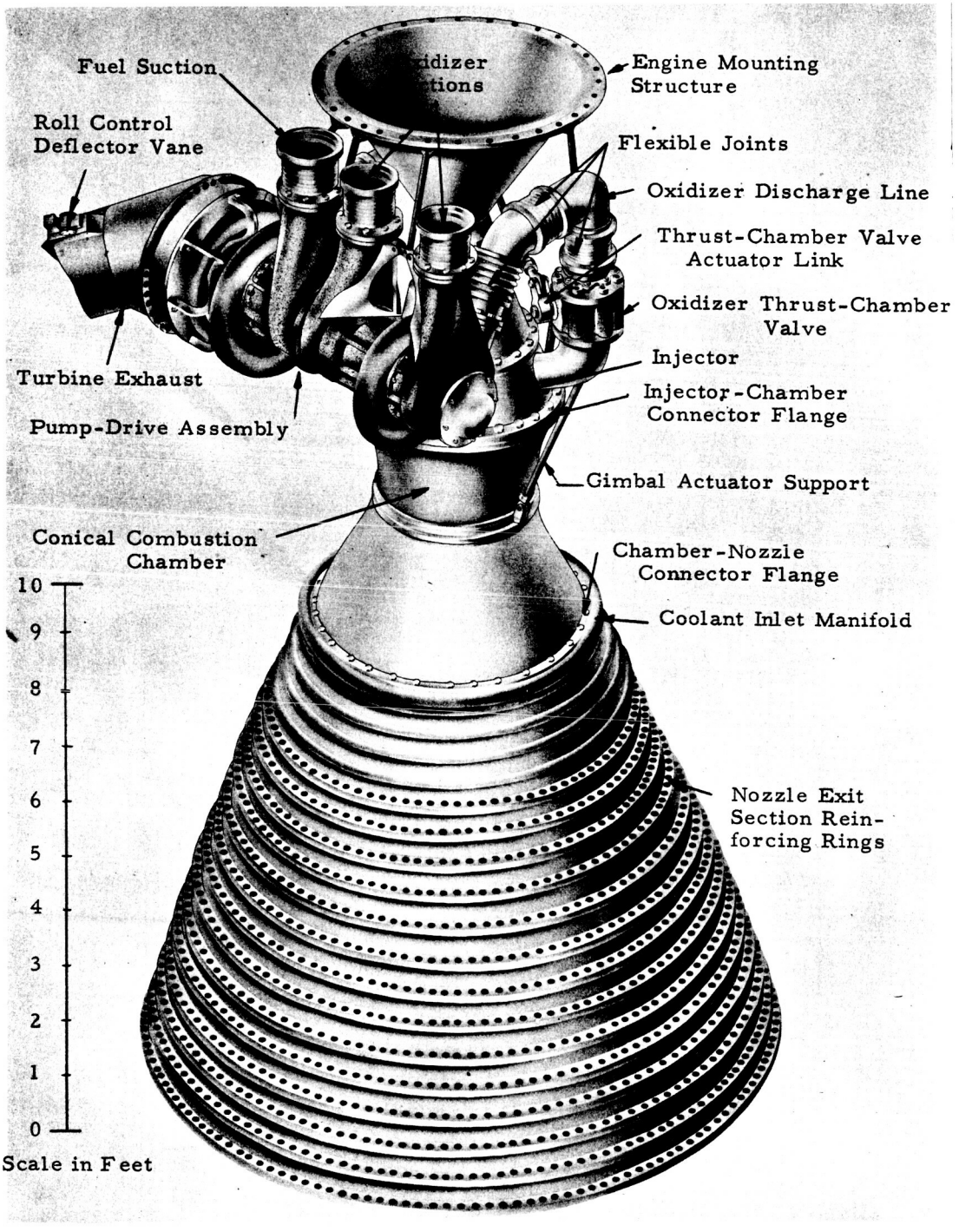
* Attribute weight x comparison number

CONSOLIDATED PARTS LIST FOR RELIABILITY COMPARISON

Frame

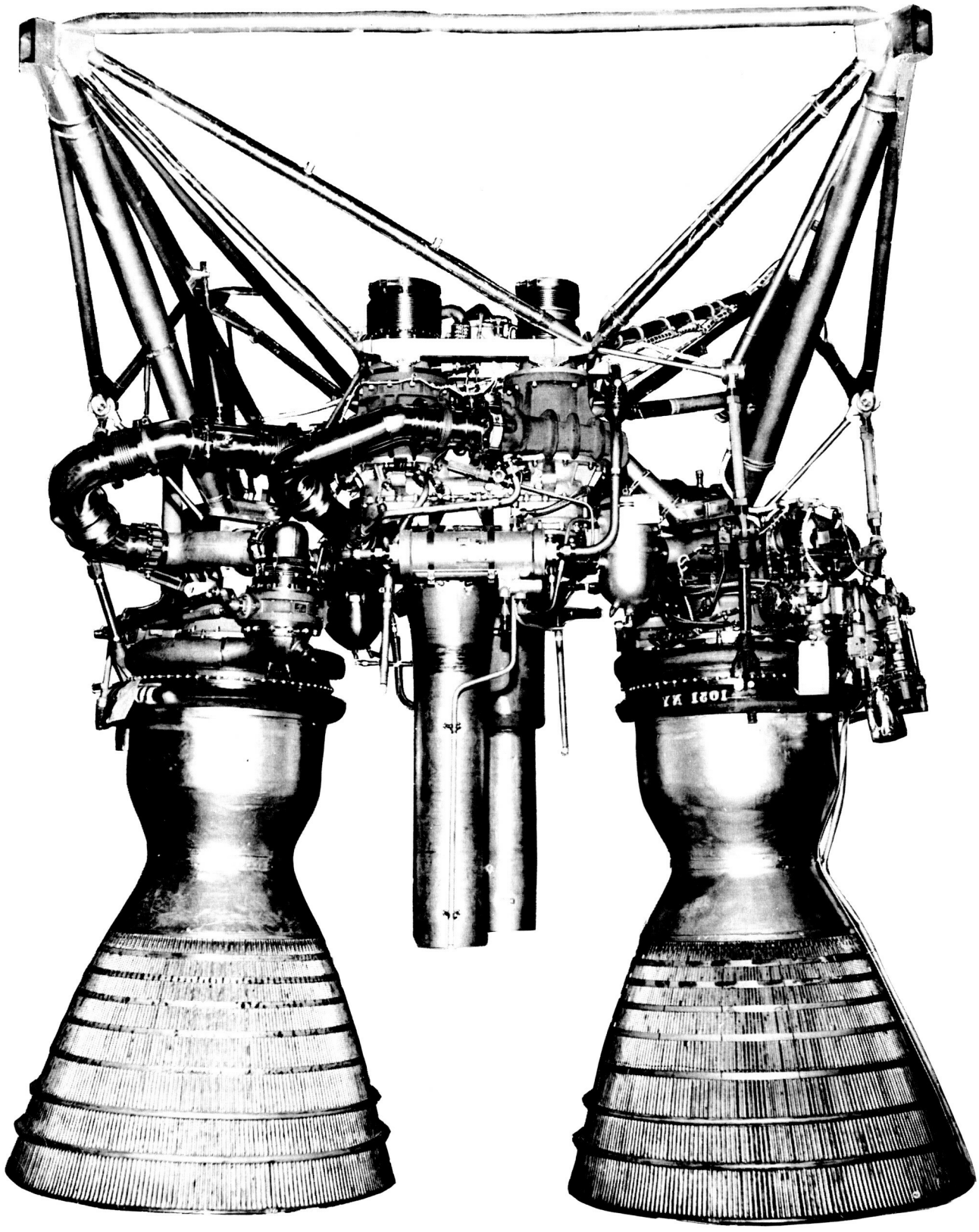
Item/Engine	YLR87-AJ-3	HS-10	3200K-4	3200K-8
Frame	1	1	1	1
Propellant Transfer System				
Fuel suction line	2	1	0	0
Oxidizer suction line	2	1	4	4
Fuel suction bellows	2	3	0	4
Oxidizer suction bellows	2	3	0	0
Oxidizer line expansion joint	0	0	4	4
Suction line flange joints	4	2	0	0
Discharge line flange joints	4	2	0	0
Discharge lines	8	4	5	5
Turbine inlet lines	2	2	4	8
Turbine discharge lines	2	1	4	4
Turbine discharge flanges	0	1	4	4
Bootstrap lines	4	2	0	0
Turbine Drive System				
Gas generator combustor	2	1	0	0
Gas generator heat exchanger	0	0	4	8
Start cartridge	1	1	0	0
Turbine inlet manifold	2	1	1	1
Turbine exhaust manifold	2	1	0	0

Item/Engine	YLR87-AJ-3	HS-10	3200K-4	3200K-8
	Turbopump Assembly			
Turbine and shaft assembly	2	1	2	2
Bearing assembly	4	3	2	2
Impeller	4	2	2	2
Gearbox	2	0	0	0
Bearing heaters	4	2	0	0
Oil coolers	2	0	0	0
Dynamic seals (sliding)	10	2	1	1
Dynamic seals (labyrinth)	2	4	3	3
Static seals (other than weld)	8	4	4	4
Housing and frame	2	1	1	1
Oil pump	2	1	0	0
Thrust chamber valves	4	2	0	0
Suction line valves	2	2	2	2
Balancing orifices	2	1	1	1
Density compensation	2	1	0	0
Thrust vector control actuator	4	2	2	2
Turbine gas diameter	0	0	4	4
LH_2 override valve	0	0	4	4
Roll control nozzle	0	1	1	1
Roll control actuator	0	1	1	1
Bleed valves	4	2	0	0



HS-10 Rocket Engine

Figure 1



YLR87-AJ-3 Rocket Engine

Figure 2

APPENDIX F

TRAJECTORY AND VEHICLE ASSUMPTIONS

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	1
II.	MISSIONS AND TRAJECTORY	1
	A. ASSUMPTIONS	2
III.	VEHICLE ASSUMPTIONS	
	A. JETTISON WEIGHT PARAMETERS	8
	B. SPECIFIC IMPULSE	14

TABLE LIST

	TABLE NO.
Weight Summary (Oxygen/Hydrogen)	1
Inert Weight Parameters (Oxygen/Hydrogen)	2

FIGURE LIST

	FIGURE NO.
Total Mission Ideal Velocity vs Ideal Velocity Before Throttle Back	1
Parking Orbit vs Ideal Velocity Before Throttle Back	2
TPA Weight vs Pump Discharge Pressure	3
Thrust Chamber Pressure vs Pump Discharge Pressure	4
Engine Sea Level Specific Impulse vs Thrust Chamber Pressure (Oxygen/Hydrogen)	5
Engine Vacuum Specific Impulse vs Thrust Chamber Pressure (Oxygen/Hydrogen)	6
Area Ratio vs Thrust Chamber Pressure	7

I. INTRODUCTION

The trajectory method and the vehicle assumptions used to calculate payload in this report are presented in this appendix.

II. MISSIONS AND TRAJECTORY

The performance of boosters is significantly affected by the flight path selected. Once the mission has been established, this flight path can be determined within reasonable limits.

Extensive experience in trajectory analysis of rocket vehicles of a wide range of vehicle sizes, performance, and thrust levels, and with a number of stages for many space missions has shown that a realistic booster-trajectory analysis can be accomplished simply by using the following vehicle trajectory.

For all space missions, a gravity-turn, continuous-thrust trajectory to a low-altitude, circular orbit and the subsequent transfer to further missions from this low-altitude orbit yield vehicle ideal velocity very close to the minimum; therefore, maximum vehicle payload will be achieved. Thus, the specific space mission has little effect upon the first-stage flight path. Hence, the low altitude orbit, which generally varies between 50 to 100 nm for most liquid-propellant two- and three-stage vehicles, has been used to establish booster flight paths.

An exception to this trajectory determination is a rescue or intercept mission, because the main criterion with these missions is to intercept in the minimum amount of time. Missions of this nature will most likely be infrequent and have not been considered in this booster-trajectory analysis. Boosters and vehicles designed for the maximum-payload flight path could be used to accomplish the intercept mission and the performance penalty would be accepted for overall economy.

II, Missions and Trajectory (cont.)

Although the booster flight path can be assumed to be independent of the space mission, vehicle staging or weight distribution between the stages is not. Besides being dependent upon the mission, vehicle staging is determined by the number of vehicle stages and the specific impulses, thrust levels, and inert weights of the individual stages. Because it appears that the most generally used missions for the next generation of vehicles will be a 300 nm, circular earth orbit and escape, these missions were used to determine optimum vehicle staging and to compare vehicles with various booster configurations.

For single-stage, 300-nm-orbit vehicles in this report, the engine was assumed to be throttled back to one-half of its original thrust at a point in the trajectory where a minimum thrust velocity was required to orbit. This not only reduces the burnout g but also provides a better trajectory, Figures 1 and 2 show the effect that the percentage of engine throttle back and point of throttle back has upon the total mission ideal velocity requirement.

Trajectory calculations involving the numerical integration of the equations of motion along the vehicle flight path were made with an IBM-7090 digital computer. The assumptions and equations used in the computer program are discussed in the following sections.

A. ASSUMPTIONS

1. Nature of the Earth

a. The earth is a sphere of radius $r_e = 2.09029 \times 10^7$ ft, rotating about the polar axis with a constant angular velocity $w_e = 7.29211 \times 10^{-5}$ rad/sec.

Throttling back 50% at 20,000 ft/sec of ideal velocity results in an approximately minimum ideal velocity requirement and keeps the parking orbit within a reasonable limit. Throttling back less than 50% increases

II, A, Assumptions (cont.)

the g's to more than 6.0 at burnout while in orbit. This level is considered undesirable for manned flight. With a 50% throttle back at 20,000 ft/sec, the g's throughout the flight are limited to 6.0 or less.

b. Properties of the atmosphere are as given in ARDC Model Atmosphere, AFCRC TN-56-204, December 1956. The atmosphere rotates with the earth.

c. The gravitational field of the earth is an inverse, square-force field. Gravitational acceleration (g_e) at the surface of the earth is 32.17405 ft/sec^2, equivalent numerically to the standard force-mass conversion factor (g_c). Gravitational acceleration (g) is in the direction opposite to the radius vector from the earth's center to the accelerated body, and its magnitude is

$$g = g_e \left(\frac{r_e}{r} \right)^2 \qquad \text{(Eq 1)}$$

2. Coordinate System Used

A nonrotating, coordinate system with the origin at the center of the earth is used.

3. Pattern of Guidance

a. Perfect guidance is realized.
b. The vehicle trajectory lies in a plane.
c. Launch is eastward from a latitude of 28.5° with respect to the equatorial plane, and the deviation of the relative wind velocity from this plane is negligible.
d. Staging is instantaneous.

II, A, Assumptions (cont.)

e. The angle pattern develops as follows:

(1) There is a vertical rise with a zero angle of attack until a velocity of 280 ft/sec is attained.

(2) There is a step-change in the angle of attack (kick angle) and a constant vehicle attitude thereafter until the angle of attack returns to zero. The value of the kick angle is determined so that the magnitude and direction of the velocity vector necessary for circular orbit are achieved simultaneously.

(3) There is a zero angle of attack thereafter (gravity turn).

4. Forces and Accelerations Acting Upon the Vehicle

a. The aerodynamic drag (D) is independent of the angle of attack and is directed opposite to the relative wind velocity vector.

$$D = \frac{\rho_a \quad v_{RW}^2}{2\, g_c} C_{D1} S_1 \qquad \text{(Eq 2)}$$

where ρ_a is the atmospheric density, v_{RW} is scalar relative wind velocity, C_{D1} is the drag coefficient for the entire vehicle, and S_1 is the cross-sectional area of the first stage, $v_{RW} = v - w_e\, r \cos\lambda$, where λ is the latitude at launch, and r is the distance from the earth's center to the vehicle. Because C_{D2} (after staging) is typically higher than C_{D1} by roughly the ratio of S_1 and S_2, and the drag during second-stage operation is small, the drag formula (Eq 2) is used through both first- and second-stage operation until second-stage burnout. The required program input is drag coefficient as a function of Mach number.

II, A, Assumptions (cont.)

b. The aerodynamic lift is zero.

c. After burnout of the second-stage, the vehicle motion is in a vacuum.

d. The instantaneous mass (m) of the vehicle in any particular stage of operation is

$$m = m_o - \dot{w}_{ENG}(t - t_o) \qquad \text{(Eq 3)}$$

where m_o is the vehicle mass at start of the stage; $\dot{w}_{ENG}$ is the propellant mass flow rate; t is the elapsed time from launch; and t_o is the elapsed time from launch until start of the stage.

e. The engine thrust (F_{ENG}) is in the direction of the longitudinal axis of the vehicle. For De Laval nozzles, its magnitude is determined from:

$$F_{ENG} = F_{ENG(vac)} - P_a A_e \qquad \text{(Eq 4)}$$

where $F_{ENG(vac)}$ is the thrust in vacuum of the engine ($\dot{w}_{ENG}$ assumed constant), P_a is the atmospheric pressure, and A_e is the effective exhaust area of the engine (the actual thrust-chamber exit area plus the projected turbine-exhaust exit area). For altitude compensating nozzles, thrust is input as a function of the atmospheric pressure.

f. The vector acceleration (a) of the vehicle is

$$\vec{a} = \frac{g_c}{m}(\vec{F}_{ENG} + \vec{D}) + g \qquad \text{(Eq 5)}$$

scalar components tangent (subscript t) and normal (subscript n) to the flight

II, A, Assumptions (cont.)

$$a_t = \dot{v} = \frac{g_c}{m}(F_{ENG}\cos\alpha - D) - g_e\left(\frac{r_e}{r}\right)^2 \cos\gamma \qquad \text{(Eq 6)}$$

$$a_n = v\dot{\Psi} = \frac{g_c}{m} F_{ENG}\sin\alpha - g_e\left(\frac{r_e}{r}\right)\sin\gamma \qquad \text{(Eq 7)}$$

The overhead arrow denotes a vector and the overhead dot denotes differentiation with respect to time. The angles are related:

$$\beta + \gamma + \Psi = \frac{\pi}{2} \qquad \text{(Eq 8)}$$

$$\theta - \alpha = \Psi \qquad \text{(Eq 9)}$$

Components of velocity tangent and normal to the earth's surface are:

$$\dot{r} = v\cos\gamma \qquad \text{(Eq 10)}$$

$$r\dot{\beta} = v\sin\gamma \qquad \text{(Eq 11)}$$

See the following list for remaining symbols.

List of Symbols

Symbol	Definition	Typical Unit
A_e	Effective exhaust area of engine (the actual thrust-chamber exit area plus the projected turbine exhaust exit area)	ft^2
a	Vehicle acceleration	ft/sec^2
a_n	Component of vehicle acceleration normal to flight path	ft/sec^2
a_t	Component of vehicle acceleration tangent to flight path	ft/sec^2

II, A, Assumptions (cont.)

List of Symbols (cont.)

Symbol	Definition	Typical Unit
C_{D1}	Drag coefficient for the entire vehicle (to be used with first-stage, cross-sectional area)	-
D	Aerodynamic drag force	lbf
F_{ENG}	Engine thrust	lbf
$F_{ENG\ (vac)}$	Engine thrust in vacuum	lbf
g	Gravitational acceleration	ft/sec^2
g_c	Conversion factor in Newton's Law	$\frac{lbm/ft}{lbf/sec^2}$
g_e	Gravitational acceleration at the earth's surface	ft/sec^2
L	Aerodynamic lift force	lbf
m	Instantaneous mass of vehicle	lbm
m_o	Vehicle mass at start of a particular stage	lbm
P_a	Atmospheric pressure	lbf/ft^2
r	Distance from earth's center to an accelerated body	ft
r_e	Radius of the earth	ft
S_1	Cross-sectional area of the first stage of the vehicle	ft^2
t	Elapsed time from launch	sec
t_o	Elapsed time from launch start of a particular stage	sec
v	Vehicle velocity	ft/sec
$\dot{w}$	Propellant mass flow rate	lbm/sec
α	Angle of attack (the angle from the instantaneous velocity vector to the longitudinal axis of the vehicle)	-

II, A, Assumptions (cont.)

List of Symbols (cont.)

Symbol	Definition	Typical Unit
β	Angle from the vertical at launch to the instantaneous radius vector from earth's center to the vehicle (the "local" vertical)	-
γ	Angle from instantaneous radius vector from earth's center to the vehicle (the local vertical to the instantaneous velocity vector)	-
θ	Angle from the horizontal at launch to the longitudinal axis of the vehicle	-
Ψ	Angle from the horizontal at launch to the instantaneous velocity vector	-
w_e	Earth's rotation about the polar axis	rad/sec
v_{RW}	Scalar relative wind velocity	ft/sec
λ	Latitude at launch	-
ρ_a	Atmospheric density	lbm/ft^3

III. VEHICLE ASSUMPTIONS

A. JETTISON WEIGHT PARAMETERS

To calculate vehicle payload, it was necessary to determine certain stage-jettison-weight parameters. A total-stage jettison weight, MD, is defined by the following relationship:

$$MD = MT + MOUT + MAF$$

where:

- MD = Total-stage jettison weight (lb)
- MT = Thrust-dependent inert weight (lb)
- MOUT = Propellant outage (lb)
- MAF = Volume dependent airframe weight (lb)
- MT = (KT) (FV)

III, A, Jettison Weight Parameters (cont.)

where:
- FV = Engine vacuum thrust (lb)
- KT = Thrust-dependent inert weight factor
- MOUT = (KOUT) (MP)

where:
- MP = Weight of usable propellant (lb)
- KOUT = Propellant outage fraction
- MAF = (KAF) (VPT)

where:
- VPT = Total propellant tank volume (ft^3)
- KAF = Propellant tank weight factor (lb of tankage/cu ft of total propellant tank volume) (lb/ft^3)
- VPT = (1 + KOUT + KULL) VP

where:
- VP = Volume of usable propellant (ft^3)
- KULL = Ullage factor

If the stage-jettison weight parameters, KT, KAF, KOUT, and KULL are known, the stage payload can be calculated from the stage-burnout weight (stage-lightoff weight less usable propellant) by using the following relationship:

MPL = MBO - MD

where:
- MPL = Payload (lb)
- MBO = Burnout weight (lb)
- MD = Jettison weight (lb)

III, A, Jettison Weight Parameters (cont.)

Values for the stage-jettison weight parameters were determined upon the basis of vehicle designs at the two-million lb thrust level. A weight breakdown of a booster stage, an upper stage, and a single stage is presented in Table 1 for oxygen/hydrogen vehicle as well as De Laval and forced-deflection nozzles. The first stage engine sea level thrust was fixed at two-million lb with a chamber pressure of 1,000 psia. All of the upper-stage weights were scaled from the booster stage weights. Scaling parameters are shown for each weight component in Table 1. These scaling parameters are only approximations but give acceptable results when the size of the vehicle to be scaled is not considerably different than the vehicle from which it is scaled.

The airframe for all the stages is of the same configuration. It consists of common bulkhead cylindrical tanks with elliptical upper and middle bulkheads, and a conical thrust transmitting structure in the lower bulkhead. Cylindrical sections are of skin-frame-stringer construction. Tank length to diameter ratios are 3:1 for the booster and upper stages, and 4:1 for the single stage. The fuel tank gas pressure is 26 psia and the oxidizer tank gas pressure is 25 psia. Tank and structure material is 2014-T6 aluminum.

Design of each stage was based upon man-rated criteria. These include: a safety factor of 1.5 in the structural design; near-neutral aerodynamic stability; and a fully loaded, free standing capability.

Inert weight factors, KAF and KT, for each stage are presented in Table 2. Inert weights from Table 1 were separated into weights that vary with tank volume and those that vary with vacuum thrust based upon the scaling factors. The sum of the volume-sensitive weights divided by the propellant tank volume equals KAF, and the sum of the thrust-sensitive weights divided by the stage vacuum thrust equals KT. Another stage parameter, KOUT, which is the propellant and gaseous residuals divided by the useful propellant weight, was calculated for each of the stages.

III, A, Jettison Weight Parameters (cont.)

Similar methods were used to establish the inert weight parameters for oxygen/hydrocarbon vehicles. Values of the stage-jettison weight parameters are summarized in the following tables.

1. Single-Stage Oxygen/Hydrogen Vehicle

Thrust-dependent inert weight factor, KT	
De Laval nozzle	0.0148
Altitude compensating nozzle	0.0143
Propellant-tank weight factor, KAF	0.626
Propellant-outage factor, KOUT	0.010
Ullage factor, KULL	0.01

2. Two-Stage Oxygen/Hydrogen Vehicle

Factor	First Stage	Second Stage
Thrust-dependent insert weight, KT		
(1) De Laval nozzle	0.0160	0.0158
(2) Altitude compensating nozzle	0.0156	0.0154
Propellant-tank weight, KAF	0.617	0.677
Propellant outage, KOUT	0.0108	0.009
Ullage, KULL	0.01	0.01

III, A, Jettison Weight Parameters (cont.)

3. Single-Stage Hydrocarbon Vehicle

Thrust-dependent, inert weight factor, KT	0.0134
Propellant-tank weight factor, KAF	0.814
Propellant-outage factor, KOUT	0.01
Ullage factor, KULL	0.01

4. Two-Stage Hydrocarbon Vehicle

Factor	First Stage	Second Stage
Thrust-dependent inert weight factor, KT	0.0147	0.014
Propellant-tank weight factor, KAF	0.813	0.914
Propellant-outage factor, KOUT	0.01	0.01
Ullage factor, KULL	0.01	0.01

The thrust-dependent weight factors were determined for an engine operating at a 1,000-psia thrust chamber pressure with a gas generator cycle. These thrust-dependent weight factors were adjusted for turbopump assembly weight increases with higher pump discharge pressure requirements for the staged combustion engine cycle and for increases in thrust chamber pressure. The variation of turbopump assembly wet weight and thrust chamber pressure with the pump discharge pressure used is shown in Figures 3 and 4 respectively. Thrust chamber assembly weight was assumed to be constant with thrust chamber pressure.

III, A, Jettison Weight Parameters (cont.)

The thrust-dependent inert-weight parameter includes engine weight, thrust-dependent structure weight, and for first stages, a weight for an engine skirt used to support the vehicle on the launching pad is included. Engine skirt weight is the main reason for higher KT factors in first stages than in upper stages. The KT factor is also lower for single-stage vehicles than for the first stage of multistage vehicles, because the interstage structure is not required.

The thrust-dependent factor for oxygen/hydrogen vehicles is larger than for vehicles using hydrocarbon propellants, because of the larger vehicle diameter required for oxygen/hydrogen propellants. A larger diameter results in larger thrust components and a heavier structure.

The propellant-tank weight factor includes all airframe weights that vary with propellant-tank volume. A safety factor of 1.5 was used in calculating the tank weights from the various tank designs. Upper-stage, oxygen/hydrogen-tank weight factors are higher than oxygen/hydrogen first-stage tank factors largely because of the insulation weight of the hydrogen tank. This insulation weight does not vary directly with propellant-tank volume, but becomes lighter per cubic foot of volume as tank size is increased.

Tank weight factors for hydrocarbons are larger than for oxygen/hydrogen, mainly because of the larger propellant head caused by the higher density of the fuel.

Propellant-outage factors are shown as approximately 0.01. This factor can be realized if a propellant utilization system is used and if approximately 80% of the propellant normally left in the propellant suction lines

III, A, Jettison Weight Parameters (cont.)

can be utilized. If the propellant in the suction lines is not used before stage shutdown in a system with the oxidizer tank on top, the unused propellant would be approximately 2% of the usable propellant weight. Because this 100% increase in unused propellant appreciably lowers the payload capability of a vehicle, methods for draining the propellant suction lines were investigated and appear to be feasible. Therefore, the propellant-outage factors shown have been used throughout for vehicle payload calculations.

An ullage factor of 0.01 was also assumed. This assumption may not be reasonable for a hydrogen tank, but because the effect of ullage upon total stage-jettison mass is extremely small, any error will not appreciably affect the results. For example, increasing the ullage factor from 0.01 to 0.05 will increase the total first-stage tank volume of a two-stage oxygen/hydrogen vehicle 4%, and the total first-stage airframe weight 4%; however, the effect upon the total first-stage jettison weight is only 1% and approximately 0.25% upon the vehicle payload.

B. SPECIFIC IMPULSE

Specific impulse values used to determine vehicle performance in this report were on the basis of a shifting equilibrium flow. A value of nozzle efficiency, η_n, of 0.97 was assumed for both oxygen/hydrogen and hydrocarbon propellants. Values of combustion efficiency, η_c, of 0.98 and 0.96 were assumed for oxygen/hydrogen and hydrocarbons, respectively. Specific impulse for oxygen/hydrogen was determined for mixture ratios of 5, 6, 7, 8 and 9 to 1. The mixture ratio for hydrocarbons was selected for approximately maximum specific impulse. Thrust chamber specific impulse values were degraded for pump operation in the staged combustion cycle and for the

III, B, Specific Impulse (cont.)

low turbine exhaust performance in the conventional gas generator cycle. The method used for degrading the thrust chamber specific impulse to determine the engine specific impulse is presented in Volume II. The variation of engine specific impulse with thrust chamber pressure and engine mixture ratio used for the parameter study of the oxygen/hydrogen staged-combustion engine cycle is shown in Figure 5 and 6.

For vehicles with a De Laval nozzle, the area ratio was determined by assuming a 4 to 1 ratio of ambient to engine exit pressure at sea level. For vehicles with altitude-compensating nozzles, the nozzle exit diameter was limited by the vehicle diameter. The variation of area ratio with the thrust chamber pressure of oxygen/hydrogen propellants is shown in Figure 7.

For all upper stages, the specific impulses were determined for a conventional gas-generator cycle, a De Laval nozzle, and an area ratio of 40:1, as well as a thrust chamber pressure of a 1000 psia.

TABLE 1

WEIGHT SUMMARY (OXYGEN/HYDROGEN)

	SCALING PARAMETER	BOOSTER STAGE DeLAVAL ENGINE	UPPER STAGE DeLAVAL ENGINE	SINGLE STAGE DeLAVAL ENGINE	BOOSTER STAGE FORCED DEFLECTION ENG.	SINGLE STAGE FORCED DEFLECTION ENG.
GUIDANCE & CONTROL[2]	1	500	500	500	500	500
FUSELAGE & EQUIPMENT		46622	11,030	47,846	46,622	47,846
PROPELLANT CONTAINER		27,323	7,221	31,804	27,323	31,804
UPPER O_2 BULKHEAD	V	973	246	806		
BULKHEAD WELDMENTS & FITTINGS	V	243	62	392		
O_2 CYLINDER SKIN & STRINGERS	V	4020	1020	5,960		
INTERNAL FRAMES	V	640	162	493		
CENTER BULKHEAD	V	970	246	1,144		
H_2 CYLINDER SKIN & STRINGERS	V	11,700	2,960	12,720		
H_2 CYLINDER INSULATION[2]	$V^{1/3}$	2,060	824	2,500		
INTERNAL FRAMES	V	1,220	309	987		
LOWER BULKHEAD	V	1,200	304	926		
BULKHEAD WELDMENTS & FITTINGS	V	560	142	831		
ANTI-SLOSH BAFFLES[2]	V	737	186	995		
MISCELLANEOUS[2]	V	3000	760	4,050		
STRUCTURAL FRAME		17,604	3,019	14,347	17,604	14,347
THRUST STRUCTURE	F	3,672	0	3675		
INTERSTAGE STRUCTURE[2]	F	3260	791	0		
STIFFENERS[2]	F	822	200	822		
UPPER END RING[2]	V	472	113	472		
(2) FIELD SPLICE END RINGS[2]	F	956	232	956		
AFT TRANSITION CORRUGATION[2] OR AFT SKIRT	F	3400	734	3,400		
TANK CORNER STRUCTURE & TIE DOWN & GIMBAL FORGE	F	3,527	580	3,527		
MISCELLANEOUS[2]	F	1,495	363	1,495		
MEASURING EQUIPMENT[2]	1	500	500	500	500	500
MISCELLANEOUS[2]	F	1,195	290	1,195	1,195	1,195
PROPULSION SYSTEM & ACCESSORIES		21,985	6,170	22,384	20,967	21,366
ENGINES (WET WEIGHT)	F	16,407	4,820	16,407	15,389	15,389
PROPELLANT CONTAINER EQUIPMENT		3,760	909	4,159	3,760	4,159
H_2 SUCTION LINES & PRE-VALVES	$FV^{1/3}$	95	15	95		
O_2 SUCTION LINES & PRE-VALVES	$FV^{1/3}$	326	50	354		
VENT LINES & VALVES	V	110	26	110		
FILL LINES & VALVES	V	36	9	36		
PNEUMATIC CONTROL SYSTEM[2]	F	329	80	329		
HELIUM BOTTLES & HELIUM GAS	V	1900	481	2150		
HELIUM BOTTLE ATTACHMENTS[2]	V	236	60	267		
HELIUM SYSTEM PLUMBING, VALVES, ETC.	$FV^{1/3}$	460	71	540		
GH_2 SYSTEM PLUMBING, VALVES, ETC.	$FV^{1/3}$	110	17	120		
PROPELLANT UTILIZATION EQUIPMENT[2]	$V^{1/3}$	158	100	158		
THRUST VECTOR CONTROL SYSTEM[2]	F	1,220	296	1,220	1,220	1,220
CONTINGENCIES[2]	F	538	145	538	538	538
STAGE INERT WEIGHT =		69,107	17,700	70,130	68,089	69,712
UNUSABLE PROPELLANT & GAS RESIDUALS		9,607	1,907	9,833	9,607	9,833
TRAPPED O_2		7,414	1,360	7,414	7,414	7,414
TANKS[2]	V	2,214	560	2,214		
LINES	$FV^{1/3}$	5,200	800	5,200		
TRAPPED H_2		493	119	493	493	493
TANKS[2]	V	433	110	433		
LINES	$FV^{1/3}$	60	9	60		
GAS RESIDUALS IN O_2 TANK	V	500	126	566	500	566
GAS RESIDUALS IN H_2 TANK	V	1,200	302	1,360	1,200	1,360
USABLE PROPELLANT RESIDUALS (OUTAGE)		2,170	550	2,488	2,170	2,488
O_2	V	1,844	467	2,080		
H_2	V	326	83	368		
STAGE RESIDUAL WEIGHT =		11,777	2,457	12,281	11,777	12,281
EXPECTED PROPELLANT CONSUMPTION		1,085,000	274,000	1,224,000	1,085,000	1224,000
STAGE WEIGHT =		1,165,884	294,157	1,307,011	1,164,866	1,305,993

NOTE: (1) THESE WEIGHTS WERE ASSUMED TO BE THRUST SENSITIVE FOR THE INERT WEIGHT PARAMETER CALCULATIONS.
(2) WEIGHTS SCALED FROM OTHER SOURCES.
(3) O_2/H_2 PROPELLANTS

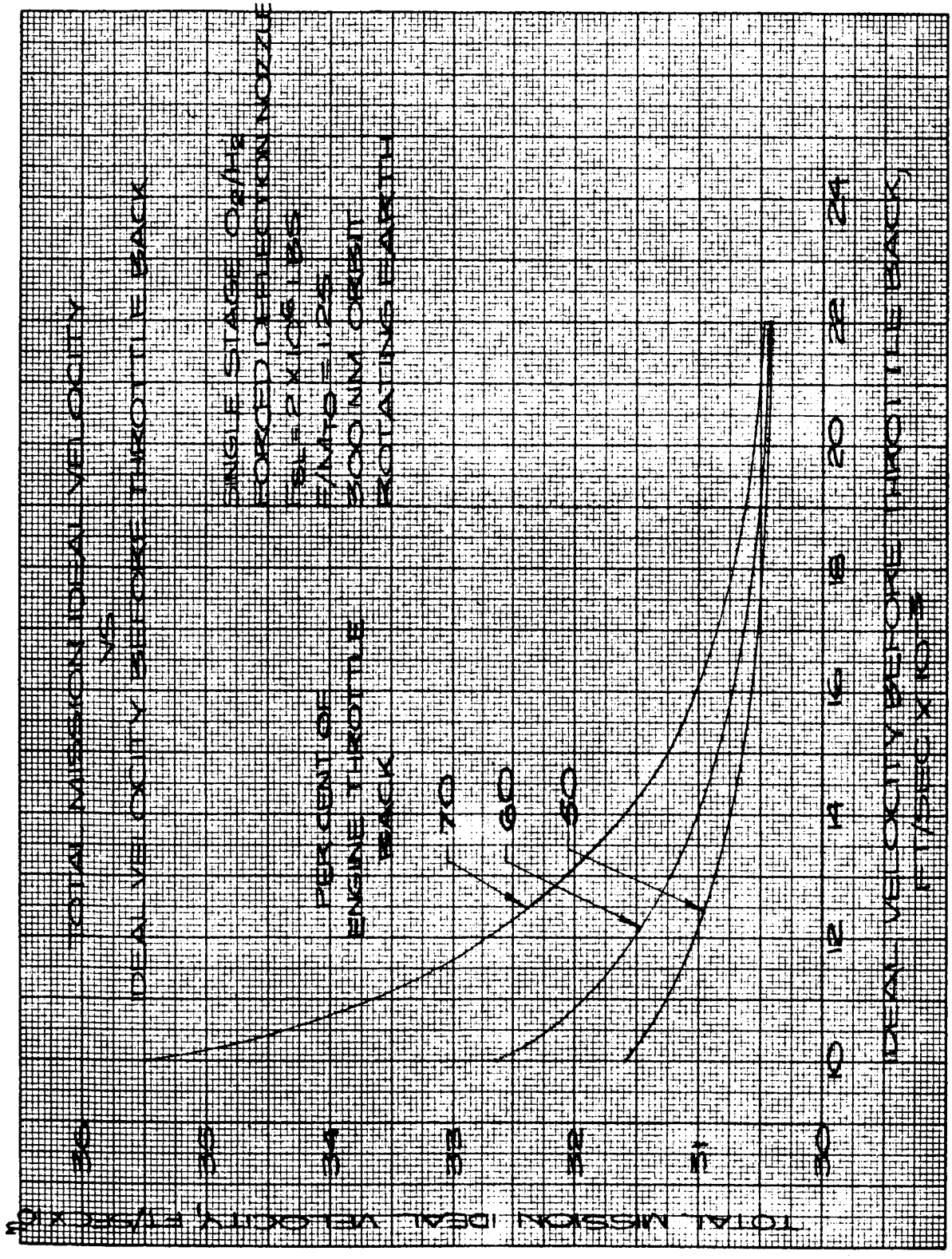
CONFIDENTIAL

TABLE 2

INERT WEIGHT PARAMETERS (OXYGEN/HYDROGEN)

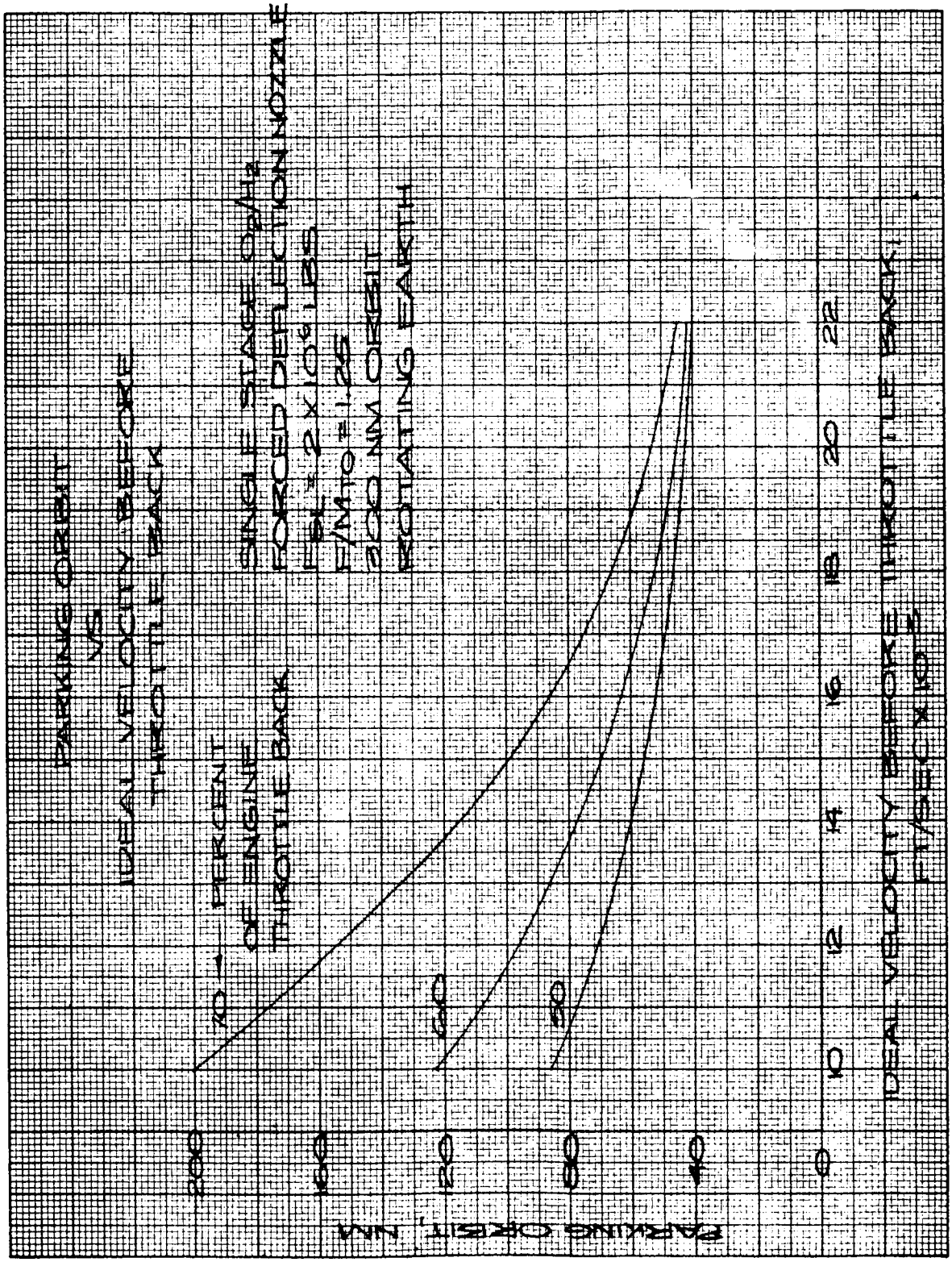
	BOOSTER STAGE DELAVAL ENG.	UPPER STAGE DELAVAL ENG.	SINGLE STAGE DELAVAL ENG.	BOOSTER STAGE FORCED DEFLECTION ENG	SINGLE STAGE FORCED DEFLECTION ENG.
STAGE VACUUM THRUST	2.39×10^6 LB	581,000 LB	2.39×10^6 LB	2.4×10^6 LB	2.4×10^6 LB
STAGE PROPELLANT TANK VOLUME	49,850 FT^3	12,570 FT^3	56,760 FT^3	49,850 FT^3	56,760 FT^3
WEIGHT BREAKDOWN					
(1) V DEPENDENT	30,235	8,016	34,997	30,235	34,997
(2) F DEPENDENT	37,872	8,684	34,733	36,854	33,715
(3) USABLE RESIDUALS (OUTAGE)	2,170	550	2,488	2,170	2,488
(4) UNUSABLE RESIDUALS	9,607	1,907	9,833	9,607	9,833
(5) INSTRUMENTATION	1,000	1,000	1,000	1,000	1,000
STAGE BURNOUT WEIGHT =	80,884	20,157	83,051	79,866	82,033
$KAF = \frac{(1) + \frac{(5)}{2}}{V_{PT}}$	$\frac{30,785}{49,850} = .617$	$\frac{8516}{12,570} = .677$	$\frac{35,497}{56,760} = .626$	$\frac{30,735}{49,850} = .617$	$\frac{35,497}{56,760} = .626$
$KT = \frac{(2) + \frac{(5)}{2}}{F_V}$	$\frac{38,372}{2.39 \times 10^6} = .0160$	$\frac{9,184}{581,000} = .0158$	$\frac{35,233}{2.39 \times 10^6} = .0148$	$\frac{37,354}{2.4 \times 10^6} = .0156$	$\frac{34,215}{2.4 \times 10^6} = .0143$
$KOUT = \frac{(3) + (4)}{M_{PROP}}$	$\frac{11,777}{1,085,000} = .0108$	$\frac{2457}{274,000} = .009$	$\frac{12,281}{1224,000} = .010$	$= .0108$	$= .010$
λ	.931	.931	936	.931	.937

Table 2



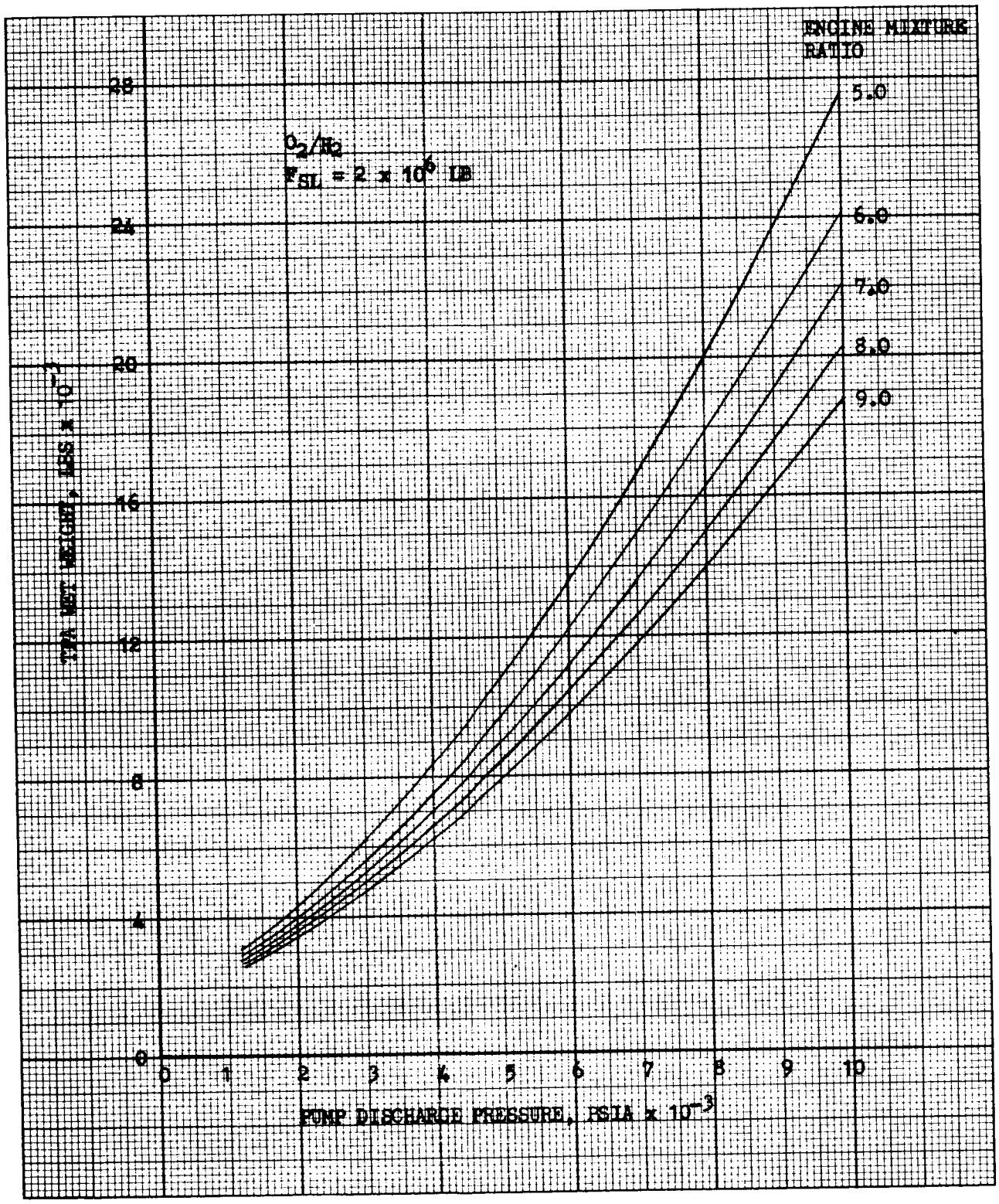
Total Mission Ideal Velocity vs Ideal Velocity Before Throttle Back

Figure 1



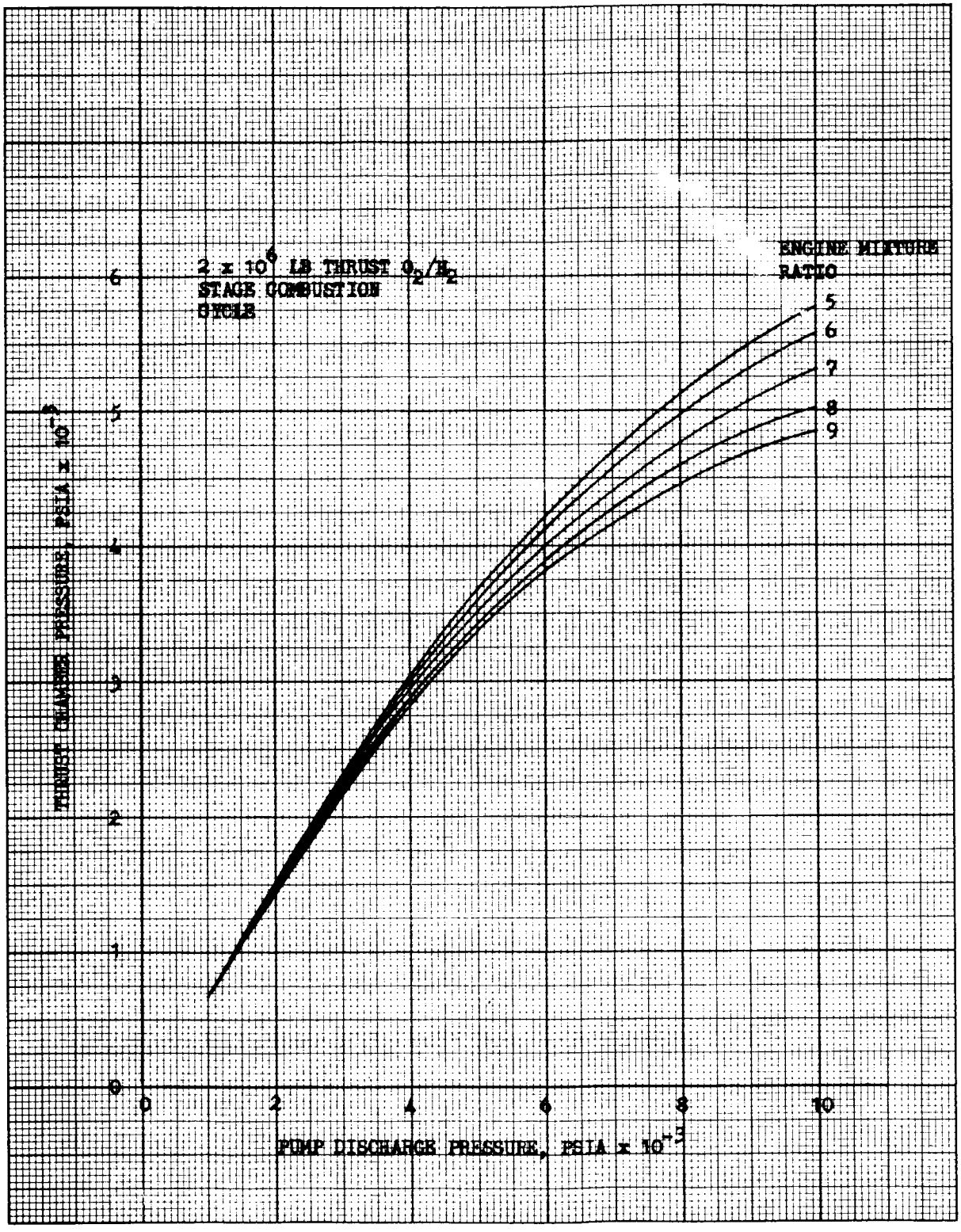
Parking Orbit vs Ideal Velocity Before Throttle Back

Figure 2



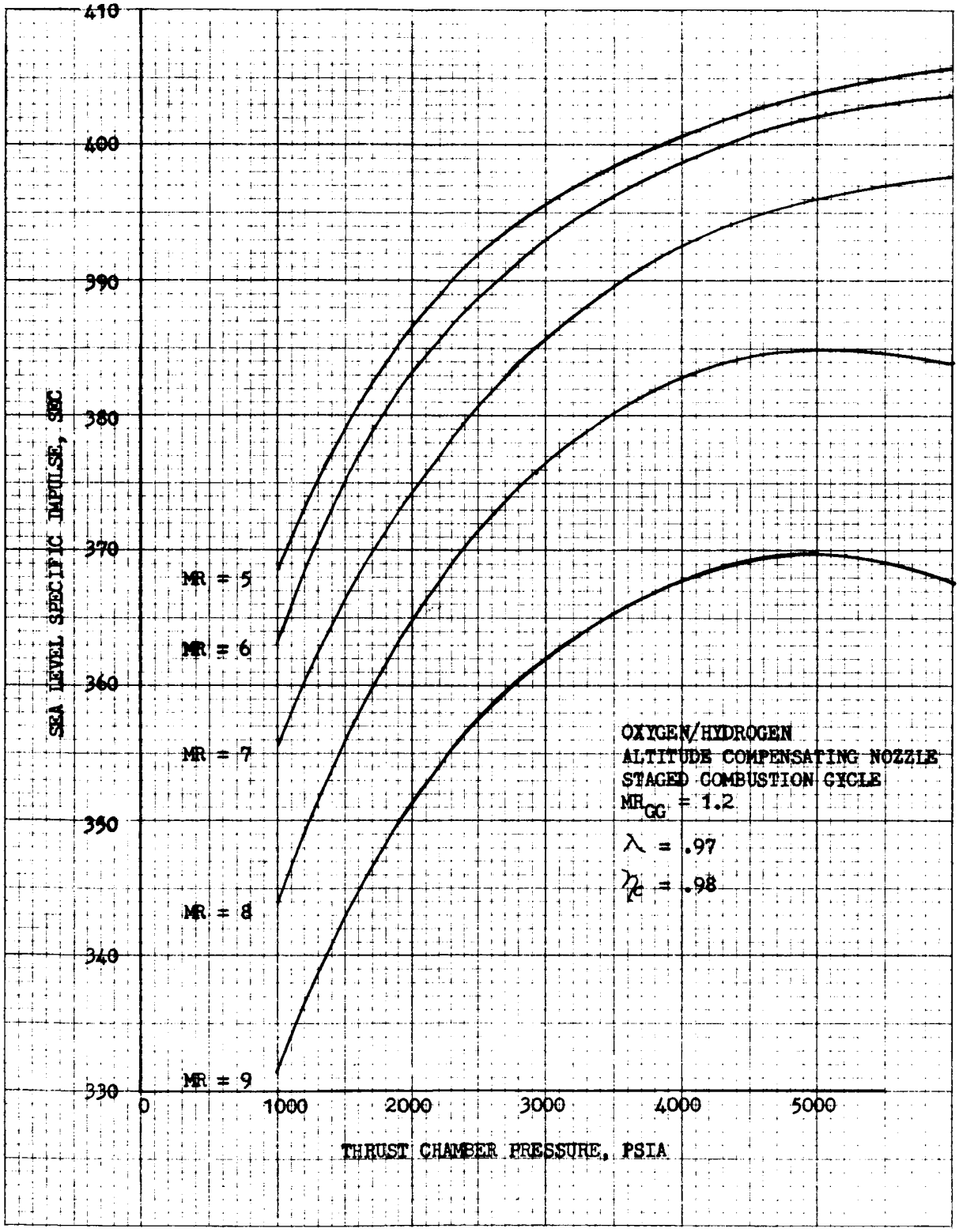
TPA Weight vs Pump Discharge Pressure

Figure 3



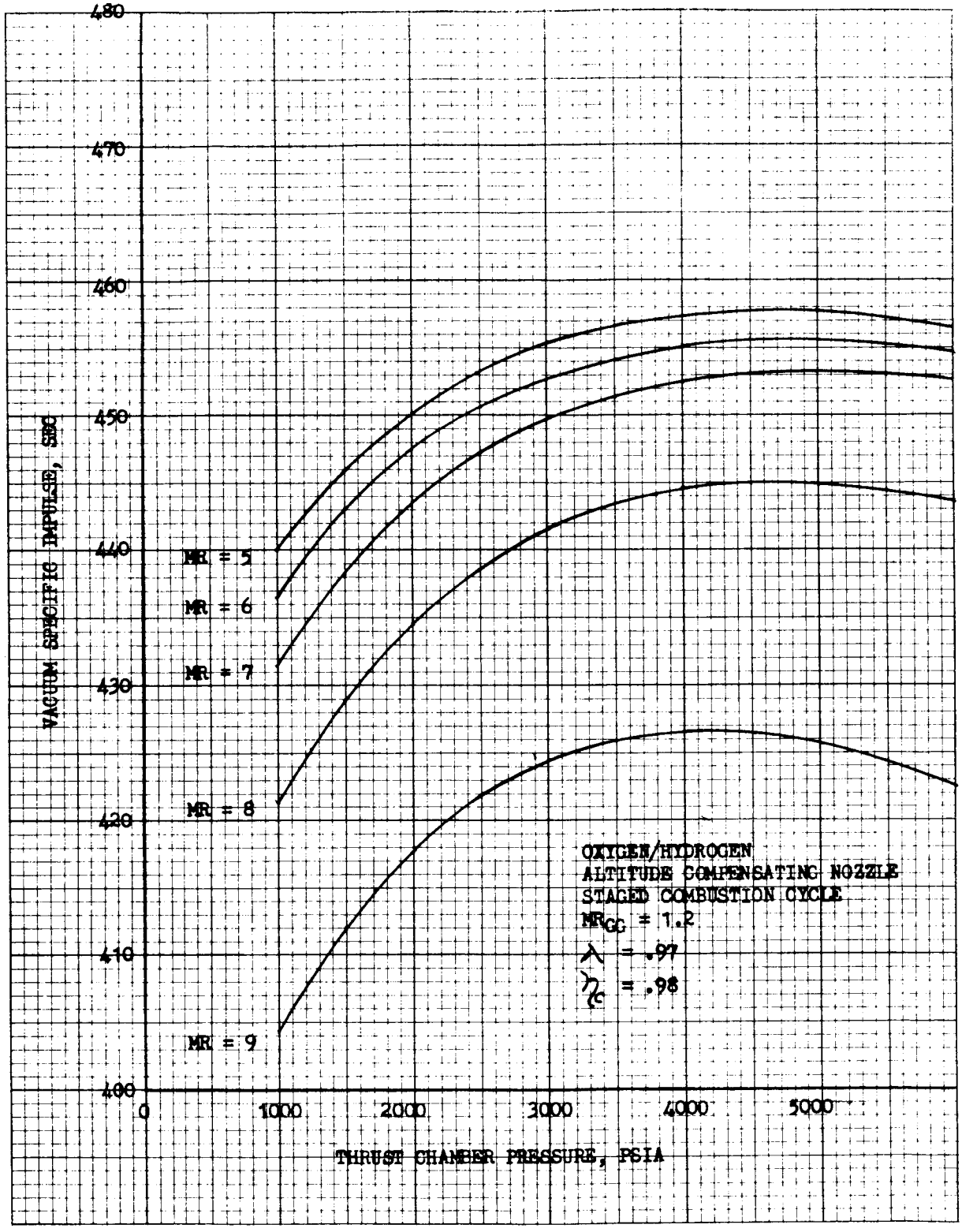
Thrust Chamber Pressure vs Pump Discharge Pressure

Figure 4



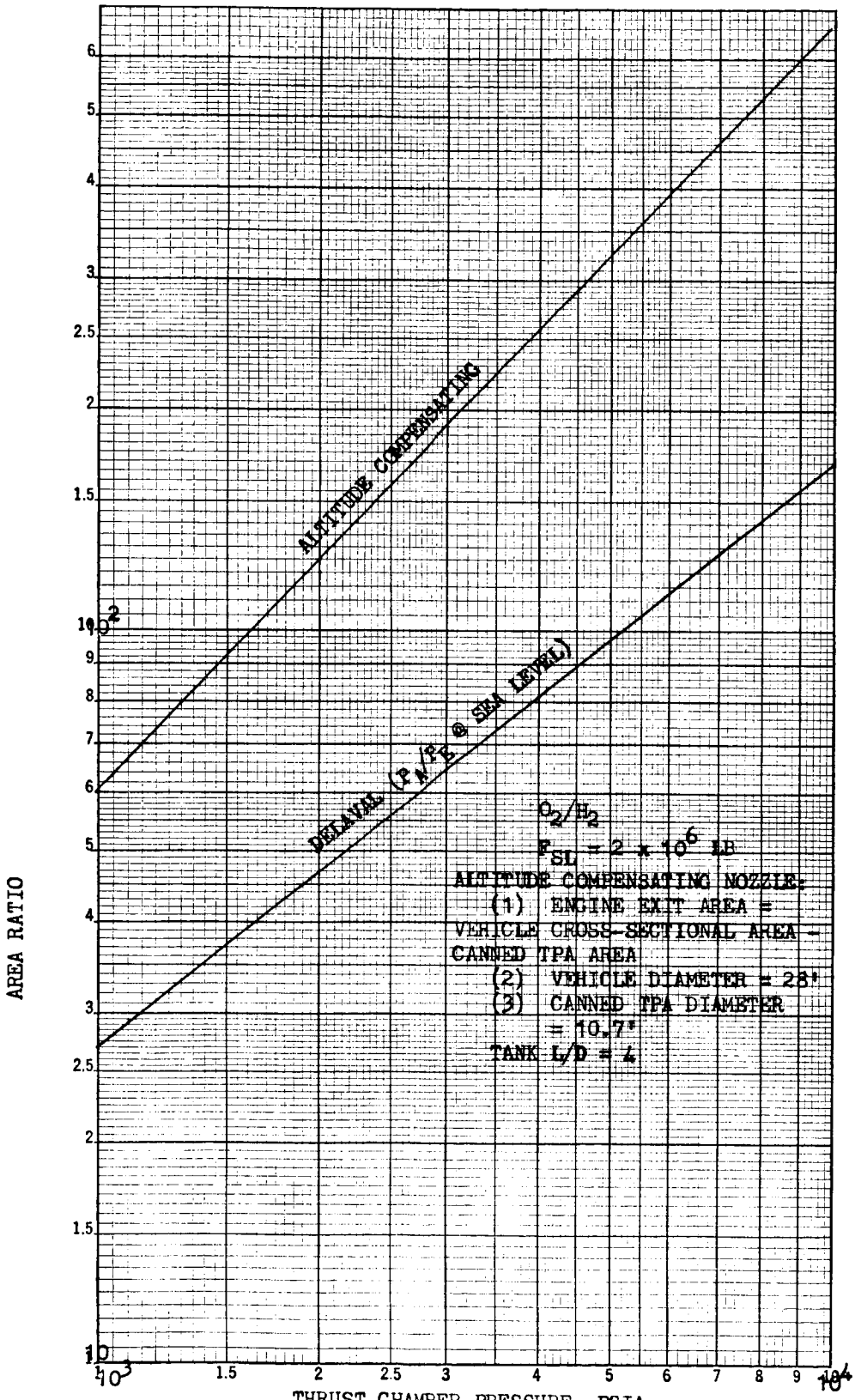
Engine Sea-Level Specific Impulse vs Thrust Chamber Pressure
(O_2H_2, Staged Combustion Cycle)

Figure 5



Engine Vacuum Specific Impulse vs Thrust Chamber Pressure
(O_2H_2, Staged Combustion Cycle)

Figure 6



Area Ratio vs Thrust Chamber Pressure

Figure 7

APPENDIX G

STEADY-STATE ANALYSIS OF A SIX-MILLION-POUND THRUST LIQUID OXYGEN/LIQUID HYDROGEN BOOSTER ENGINE

TABLE OF CONTENTS

		PAGE NO.
I.	OBJECTIVE	I-1
II.	DEFINITION OF BASIC ENGINE SYSTEM	II-1
III.	SUMMARY OF RESULTS	III-1
IV.	ASSUMPTIONS	IV-1
	A. MISSILE ACCELERATION AND FLIGHT DURATION	IV-1
	B. PROPELLANT TANKAGE CONFIGURATION	IV-1
	C. TANKAGE LOADING TOLERANCES	IV-2
	D. ENGINE TOLERANCES	IV-2
	E. ACCURACY OF MONITORED SIGNALS	IV-3
V.	STEADY-STATE ANALYSIS	V-1
	A. METHOD OF ANALYSIS	V-1
	B. ENGINE PERFORMANCE--NO ENGINE CONTROL VALVES	V-1
	C. ENGINE PERFORMANCE PROPELLANT UTILIZATION CONTROL	V-5
	D. ENGINE PERFORMANCE--THRUST CONTROL	V-8

TABLE LIST

	TABLE NO.
Performance and Operating Parameters--Engine Balanced for 6-Million-lb Takeoff Thrust	1
Performance and Operating Parameters--Engine Performance at Static Condition	2
Effect of Engine Tolerances upon Engine Parameters	3
Effect of Engine Tolerances upon Engine Parameters at Constant Mixture Ratio	4
Effect of Engine Tolerances upon Engine Parameters--Volume-Measuring Propellant Utilization System	5
Effect of Engine Tolerances upon Engine Parameters at Constant Thrust Control	6

FIGURE LIST

	FIGURE NO.
Engine System Schematic	1
Longitudinal Acceleration vs Time--6-Million-lb-Thrust O_2/H_2 Booster	2
Propellant Static Heads--Oxidizer Tank on Top	3
Propellant Static Heads--Fuel Tank on Top	4
Total Pump Suction Pressures--Oxidizer Tank on Top	5
Total Pump Suction Pressures--Fuel Tank on Top	6
Thrust Variation During Flight--No Variable Controls	7
Mixture Ratio Variation During Flight--No Variable Controls	8
Engine I_{sp} Variation During Flight--No Variable Controls	9
Chamber Pressure Variation During Flight--No Variable Controls	10

FIGURE LIST (cont.)

	FIGURE NO.
Resistance Coefficients Required for Engine Balance in the Oxidizer Line to the Oxidizer Turbopump Gas Generator	11
Resistance Coefficients Required for Engine Balance in the Fuel Line to the Oxidizer Turbopump Gas Generator	12
Resistance Coefficients Required for Engine Balance in the Oxidizer Line to the Fuel Turbopump Gas Generator	13
Resistance Coefficients Required for Engine Balance in the Fuel Line to the Fuel Turbopump Gas Generator	14
Engine Thrust Variation During Flight with Constant Mixture Ratio Control	15
Engine I_{sp} Variation During Flight with Constant Mixture Ratio Control	16
Chamber Pressure Variation During Flight with Constant Mixture Ratio Control	17
Engine Thrust at Off-Design Mixture Ratio Conditions	18
Chamber Pressure at Off-Design Mixture Ratio Conditions	19
Engine I_{sp} at Off-Design Mixture Ratio Conditions	20
Engine Mixture Ratio for Engine Operation at Constant Chamber Pressure	21
Engine I_{sp} for Engine Operation at Constant Chamber Pressure	22

I. OBJECTIVE

The objective of this investigation was to analyze the steady-state performance characteristics of a six-million-pound LOX/LH_2 booster engine. The elements to be evaluated included engine balance point, effect of system tolerances upon engine performance, effects of missile acceleration upon engine performance, propellant utilization systems, and thrust control systems.

II. DEFINITION OF BASIC ENGINE SYSTEM

The control system investigation was predicated on a given basic engine system, selected from previous analysis results and design considerations of engine components at the six-million lb thrust level size. Elements of this engine system include:

A. Engine thrust: Six-million-pound thrust.

B. Propellants: Liquid oxygen and liquid hydrogen

C. Thrust chamber: Single, regeneratively-cooled chamber, conventional multihole injector.

D. Turbopumps: Single and separate oxidizer and fuel turbopump assemblies, each with integral multiple gas generators.

E. Minimum control: Main propellant valves, balancing orifices.

F. Turbopump power source: Individual gas generators for each turbopump, with propellants tapped from the engine discharge lines upstream of the main propellant valves.

An engine system schematic is shown as Figure 1.

III. SUMMARY OF RESULTS

Engine performance during powered flight was determined and analyzed for engine systems balanced for a takeoff thrust of six-million-pounds, systems having propellant utilization provisions, and systems having thrust controllers. If no variable controls are used, engine thrust and mixture ratio during powered flight may vary up to approximately + 11%. Knowing the vehicle acceleration characteristics, residual propellant at burnout can be minimized by loading 1.4% extra fuel, for which the greatest possible residual propellant weight is equivalent to 3.3% loaded fuel weight.

If the propulsion system is to have a propellant utilization system, engine performance variations are best minimized by controlling the resistance in the propellant lines to the oxidizer turbopump gas generator. If the propellant utilization system is the type which transmits a continuous signal describing tankage propellant mixture ratio, engine thrust can be maintained within $^{+1.1}_{-2.8}$% of design thrust. Use of a volume measuring propellant utilization system reduces this thrust variation to $^{+1.1}_{-1.2}$%. If a propellant utilization signal is transmitted periodically only, at times the engine will operate at an off-mixture ratio condition, but will operate satisfactorily up to $\pm$ 10% of the design mixture ratio. Additional thrust variation for these conditions is approximately one- and one-half times greater than the percentage variation of mixture ratio from its design value.

Engine performance, including engine mixture ratio, can be closely controlled by monitoring thrust chamber pressure instead of mixture ratio. With this control, chamber pressure is theoretically maintained constant, and engine mixture ratio variation is held within $^{+1.8}_{-0.7}$% of nominal. While chamber pressure control does not give perfect propellant utilization, it still might be attractive because the control system required is not as complex as those for most true propellant utilization systems.

III, Summary (cont.)

Gas generator propellants were tapped from the engine discharge lines upstream of the main propellant valves in this control system investigation. However, results are equally applicable to an engine system where the propellants are tapped downstream of the main propellant valves. In the latter instance, the resistances in the propellant lines would be redistributed, but the engine controllability and sensitivity remains the same.

IV. ASSUMPTIONS

A. MISSILE ACCELERATION AND FLIGHT DURATION

To evaluate the engine performance as it would exist under flight conditions, it is necessary to know the variation of total head conditions at the engine pump inlets during powered flight. The total head is derived from the static head, which is a function of the propellant tank configuration and the acceleration of the vehicle during flight. The relationship of missile longitudinal acceleration and time, shown in Figure 2, represents a typical relationship for a six-million-pound thrust booster mission. Takeoff acceleration $(\frac{a}{g})$ was 1.25 g, burnout acceleration 4.7 g, and powered flight duration 187 sec.

B. PROPELLANT TANKAGE CONFIGURATION

The tanks assumed for this investigation were the conventional cylindrical type, having elliptical heads. They were designed using an over-all L/D ratio of 3:1, and sized for an engine mixture ratio of 5.66:1 (thrust chamber assembly mixture ratio of 6:1). Ullage and outage were assumed to be 4% and 1% respectively for the hydrogen tank, and 1% and 1% for the oxidizer tank.

Tankage configurations with the oxidizer tank located on top of the fuel tank, and vice versa, were considered. The static propellant heads during engine operation were determined for both configurations, and are shown in Figures 3 and 4. Integrating these curves with the acceleration curve, the relationships of total pump suction pressure to time-after-launch were developed, and are shown in Figures 5 and 6. A propellant tank pressurization system was included to provide a minimum pressure of 1 psi above the propellant vapor pressure.

IV, B, Propellant Tankage Configuration (cont.)

Figures 5 and 6 show that the total oxidizer and fuel suction pressure variations throughout flight, relative to both themselves and to each other, are greatest when the oxidizer tank is located on top. Therefore, engine performance and parameter variations are expected to be greater with this tankage configuration when an engine system has no compensating controls. Moreover, a greater control range is required if a propellant utilization system or thrust control system is used. Because it was desired that the results of this control system investigation be applicable to a variety of tankage configurations and suction conditions, the oxidizer-on-top tankage configuration was selected as the one providing results most representative of the possible variation limits that could be encountered on actual missiles.

C. TANKAGE LOADING TOLERANCES

It was assumed that both propellant tanks could be loaded to the desired mixture ratio $\pm$ 0.2%.

D. ENGINE TOLERANCES

The effect of internal and external tolerances upon engine performance were considered in this investigation. The following variations were included:

1. External Tolerances

   a. Oxidizer tank pressurization system pressure: $\pm$ 2 psi.

   b. Fuel tank pressurization system pressure: $\pm$ 2 psi.

IV, D, Engine Tolerances (cont.)

c. Oxidizer suction temperature: + 2°F.

d. Fuel suction temperature: + 2°F.

2. Internal Tolerances

a. Oxidizer pump head: ± 1%.

b. Fuel pump head: ± 1%.

c. Hydraulic resistances: ± 1%.

E. ACCURACY OF MONITORED SIGNALS

In the investigation of propellant utilization systems (mixture ratio control) and thrust control systems (chamber pressure control), it was assumed that the accuracy of the instrumentation used to measure these parameters and generate appropriate error signals was 100%. While this accuracy cannot be expected in actual systems, it is a valid assumption when comparing them. Thus, the variation in engine parameters determined in this investigation will be slightly smaller than those expected during actual engine operation.

V. STEADY-STATE ANALYSIS

A. METHOD OF ANALYSIS

The entire investigation was made using the Aerojet-General steady-state computer program. It was divided into three parts:

1. Engine performance with no engine controls.
2. Control systems for propellant utilization controls.
3. Control systems for thrust control.

B. ENGINE PERFORMANCE--NO ENGINE CONTROL VALVES

1. Engine Balance

Because all engine component are not perfectly matched, it is usually necessary to balance the engine by adjusting component or line resistances to operate the engine at its design conditions. With a separate turbopump and separate gas generator engine system, this can be accomplished by changing the resistance in the main propellant lines, the bootstrap propellant lines, or the hot gas lines between the gas generators and turbines. The most simple and convenient way is by orificing the bootstrap lines, and this method was used to obtain engine balance points.

The engine was balanced to give a takeoff thrust of six-million pounds. Thrust chamber pressure (plenum) and mixture ratio were 1,000 psia and 6:1, respectively. Both gas generator mixture ratios were 0.8:1. Suction pressures corresponding to the liftoff condition ($\frac{a}{g}$ = 1.25) were used. Engine performance, parameters, and pressure schedule are summarized in Table 1.

V, B, Engine Performance--No Engine Control Valves (cont.)

Maintaining the same engine balance, engine performance for the static condition after engine start, but before liftoff ($\frac{a}{g}$ = 1.0) was determined, and is summarized in Table 2.

2. Effect of Acceleration Head Variation Upon Engine Performance

With the engine balanced for a takeoff thrust of six-million pounds, the effects of the changing suction conditions (resulting from the total head change) upon engine performance were determined. Resulting engine thrust, engine I_{sp}, engine mixture ratio, and chamber pressure variations during flight are shown in Figures 7 through 10. Maximum variations are summarized as follows:

	Percent
I_{sp}	+ 0.87
M.R.	+ 6.5
F	+ 9.8
P_c	+ 7.5

The undesirable large changes in mixture ratio, thrust, and chamber pressure indicate one advantage for a fuel-tank-on-top tankage configuration, in which case these variations would have been smaller.

3. Effect of Engine Tolerances Upon Engine Performance

The variations of engine performance to be expected as a result of internal and external engine tolerances were determined. Tolerances used in this analysis are listed in Section IV, D, of this appendix. The resulting variations in thrust, engine mixture ratio, engine I_{sp}, and chamber pressure are given

V, B, Engine Performance--No Engine Control Valves (cont.)

in Table 3. In determining the total variation from all engine tolerances that can reasonably be expected with an actual engine, a root-sum-square average of all variations was used. It was concluded the following variations can be anticipated from engine tolerances:

	Percent
I_{sp}	+0.15 -0.23
M.R.	+1.5 -1.0
F	+1.1 -1.1
P_c	+0.84 -0.95

4. Total Performance Variation

The total anticipated variation in engine performance is obtained by adding the variations caused by suction condition change and the effect of engine tolerances. These variations are summarized as follows:

Variable Parameter	Engine* Thrust Variation Percent	Engine Mixture Ratio Variation Percent	Engine I_{sp}* Variation Percent	Chamber Pres. Variation Percent
Suction Conditions	+9.8	+6.6	+0.87	+7.5
Engine Tolerances	+1.1 -1.1	+1.5 -1.0	+0.15 -0.23	+0.84 -0.94
Total Variation	+10.9 -1.1	+8.1 -1.0	+1.02 -0.23	+8.3 -0.94

* All thrust and I_{sp} values corrected to sea-level conditions.

V, B, Engine Performance--No Engine Control Valves (cont.)

5. Propellant Utilization Considerations

Propellant utilization is a control placed in vehicle systems to ensure simultaneous, or near simultaneous, exhaustion of propellants in the propellant tank at engine burnout. This control is normally effected by periodically, or continually, changing engine mixture ratio so that the mean engine mixture ratio throughout the run is equal to the propellant tank loading ratio. Without control, these mixture ratios are different because of propellant loading inaccuracies, engine mixture ratio shift caused by engine tolerances, and engine mixture ratio shift from changing static and acceleration heads.

In engine systems that do not have propellant utilization provisions, propellant outage weight can be minimized by proper selection of the propellant tank loading mixture ratio. Knowing the static and acceleration heads during flight, the mean engine mixture ratio (excluding engine tolerance effects) can be calculated. This mean mixture ratio is then modified to include engine tolerance effects and loading inaccuracies (the procedure for which is discussed below) and the propellant tanks are loaded to this mixture ratio.

In propulsion systems where engine mixture ratio is high, and the fuel density is low, as is the case with LOX/LH_2 systems, minimization of residual propellant weight by loading extra fuel to ensure oxidizer exhaustion is attractive. The engine steady-state analysis indicated that engine tolerance effects may produce a mixture ratio shift of $^{+1.5}_{-1.0}$%. Using these results and an assumed $\pm$ 0.2% tank loading accuracy, it was computed that the loading of 1.4% extra fuel, or 0.21% extra total propellants, will assure oxidizer exhaustion. If the fuel tank were loaded high, oxidizer tank low, and engine mixture ratio were high, 3.3% of the fuel would be left in the tank at oxidizer exhaustion.

V, Steady-State Analysis (cont.)

C. ENGINE PERFORMANCE PROPELLANT UTILIZATION CONTROL

1. Systems Considered and Summary of Results

Several methods for controlling engine mixture ratio were considered, including throttling valves in the gas generator feed lines to the oxidizer or fuel turbopump gas generators, throttling valves in the main oxidizer or fuel propellant lines, throttling valves in the turbopump hot gas systems, and propellant bypass back to the pump suction lines.

The hot gas and propellant bypass systems are unattractive because components and associated plumbing not normally required for engine operation are required. Also, in the case of the hot gas system, the inherent problems in hot gas control systems are present. Throttling propellants in the main propellant lines could be accomplished with the main propellant valves; however, the analysis performed for the 200,000-lb thrust LOX/LH_2 engine indicates that a separate valve, located in the oxidizer line, is a better system. Again, extra components are required. The simplest system, which also has the lowest power requirements, is to regulate one or both of the gas generator valves. Therefore, it was decided to analyze this type of system first, and then, is satisfactory control could not be obtained, other systems would be investigated.

It is evident that perfect propellant utilization and thrust control could simultaneously be obtained if mixture ratio and thrust were monitored by controlling both the oxidizer and fuel gas generator valves. However, the complex control intelligence required of this system makes it unattractive. To determine the characteristics of the line resistance variations throughout flight, engine operation was analyzed for this system. The results, showing the line resistance coefficients (which are inversely proportional to the metered areas) required

V, C, Engine Performance Propellant Utilization Control (cont.)

throughout flight, are given in Figures 11 through 14. It is noted that while the oxidizer turbopump gas generator line resistances changed considerably during the flight, the line resistances to the fuel turbopump gas generator changed only slightly. Therefore, it was decided to evaluate a system with mixture ratio monitored and controlling the oxidizer turbopump gas generator valve only.

The results of analysis, discussed in detail below, showed that satisfactory control can be obtained with the single gas generator valve controller. It was decided to select this system without further analysis of the other systems considered.

2. Engine Operation at Controlled Mixture ratio

Using the oxidizer gas generator valve controller, engine performance at constant design mixture ratio of 5.66 (thrust chamber assembly mixture ratio of 6:1) was determined. The variation of engine thrust, engine I_{sp}, and chamber pressure throughout flight are shown in Figures 15 through 17. Engine thrust (sea-level corrected) varied $^{+0.03}_{-0.16}$% and engine I_{sp} $^{+0.038}_{-0.0}$% from their design values.

The ability of the system to correct for engine tolerances was also determined, and the results are given in Table 4. Using the controller, the maximum anticipated engine performance variations are $^{+1.0}_{-2.6}$% for engine thrust, $^{+0.17}_{-0.43}$% for engine I_{sp}, and $^{+0.9}_{-2.2}$% for chamber pressure.

If correction for loading inaccuracies is required, engine mixture ratio slightly different from the design valve would be necessary. For the limiting loading tolerances, engine mixture ratio would be 5.86 or 5.64:1. This would cause an increase in the engine thrust variation of $\pm$ 0.5%.

V, C, Engine Performance Propellant Utilization Control (cont.)

3. Engine Operation with Volume Measuring Propellant Utilization System

If the propellant utilization system is designed to measure propellant tank volume instead of propellant mass or mass flow rate, significantly less thrust variation from engine tolerances results. Engine operating characteristics for all trajectory conditions and all engine tolerance conditions, except those involving propellant density changes are independent of the type of propellant utilization system used. The volumetric change of oxidizer with slight shifts in density is small, and may be considered insignificant. However, the specific volume of hydrogen changes rapidly with small density variations; and therefore, the density change is reflected in the input signal to the gas generator valve controller when a volumetric propellant utilization control is used. Computer analysis indicates that with such a system, total thrust variation from engine tolerances is $^{+1.1}_{-1.0}$%. The complete results are given in Table 5. Integrating these variations with those from the trajectory, a total engine thrust variation of $^{+1.1}_{-1.2}$% is anticipated.

4. Engine Operation at Off-Design Mixture Ratio

Because some propellant utilization systems use only periodic mixture ratio correction signals to the engine, the engine will, at times, necessarily have to operate at mixture ratios different from the design value to obtain near simultaneous propellant exhaustion. The extent of these mixture ratio deviations depend upon the frequency of the correction signals from the propellant utilization system. Engine performance was determined up to $\pm$ 10% of design mixture ratio and the results are summarized in Figures 18 to 20.

V, C, Engine Performance Propellant Utilization Control (cont.)

At the extreme conditions, performance is as follows:

a. Engine thrust

at + 10% MR: + 15%

at - 10% MR: - 15%

b. Engine I_{sp}

at + 10% MR: + 1.0%

at - 10% MR: - 1.7%

D. ENGINE PERFORMANCE--THRUST CONTROL

In most thrust-control systems, engine thrust chamber pressure, being nearly proportional to thrust, is monitored and maintained constant at the design value by the engine control system. The control system required to maintain chamber pressure is not as complex as that required for a true propellant utilization system. Therefore, an analysis was made to determine the ability of such a system to maintain engine mixture ratio and engine I_{sp} for the varying suction conditions and engine tolerances. As in the propellant utilization system analysis, the monitored signal controlled the oxidizer turbopump gas generator valve. The results are given in Figures 21 and 22, and Table 6. Engine mixture ratio engine I_{sp} variations caused by acceleration head changes were $^{+0.10}_{-0.02}$% and + 0.05% respectively. The effect of engine tolerances caused an additional mixture ratio variation of $^{+1.8}_{-0.7}$% and an I_{sp} variation of $^{+0.83}_{-0.16}$%.

V, D, Engine Performance--Thrust Control (cont.)

Because the mixture ratio variations are small, it appears feasible to consider a thrust control system for propellant utilization. Residual propellant at burnout cannot be eliminated because mixture ratio control is not perfect and propellant loading inaccuracies cannot be compensated for. However, its weight can be minimized by loading 1.1% extra fuel, which would limit the maximum residual propellant to 3.4% of the loaded fuel.

(End Text)

TABLE 1

PERFORMANCE AND OPERATING PARAMETERS, ENGINE BALANCED FOR 6 x 10^6 LB TAKEOFF THRUST

Engine	Dimension	Sea Level Values
Ambient Pressure	psia	14.697
Acceleration a/g	— —	1.25
Instantaneous Specific Impulse (Engine)	sec	343.27
Engine Mixture Ratio	- - -	5.66
Total Engine Thrust (Excluding Turbine exhaust)	lb	6 x 10^6
TCA		
Thrust	lb	6 x 10^6
Thrust Chamber Pressure (Plenum Total)	psia	1000
Thrust Chamber Pressure (Injector face)	psia	1052
Instantaneous Specific Impulse (TC)	lb/lb/sec	349.44
Thrust Coefficient	- - -	1,437
Characteristic Velocity	ft/ sec	7,822
Effective Exhaust Velocity	ft/sec	11,243
Nozzle Exit Design Pressure	psia	5.34
Nozzle Pressure Ratio (P_{op}/P_o	- - -	187
Total Propellant Flow Rate	lb/sec	17,170
Mixture Ratio $\dot{w}_o/\dot{w}_f$	- - -	6.0
Fuel Flow Rate	lb/sec	2,453
Oxidizer Flow Rate	lb/sec	14,717
Nozzle Area Ratio	- - -	20
Nozzle Throat Area	in^2	3,968

TABLE 1 (cont.)

Turbo Pump	Dimension	Fuel	Oxidizer
Ambient Temperature	°F	60	60
Propellant Temperature			
Inlet	°F	-421	-294
Outlet	°F	-406	-294
Propellant Density			
Inlet	lb/ft^3	4.37	70.4
Outlet	lb/ft^3	4.44	70.4
Propellant Vapor Pressure	psia	20.8	19.4
Shaft Speed	rpm	8,227	2,866
Total Discharge Pressure	psia	1,308	1,279
Static Discharge Pressure	psia		
Total Suction Pressure	psia	36.6	100.4
Static Suction Pressure	psia		
Total Pressure Rise	psi	1,271.4	1,178.6
Total Head Rise	ft	41,234	2,411
Propellant Flow Rate	lb/sec	2,624	14,854
Capacity	gpm	265,257	94,695
Specific Speed	- - -	1,464	2,564
Efficiency	%	81.9	85.4
Fluid Horsepower	hp	196,741	65,110
Shaft Horsepower	hp	240,158	76,250
Net Positive Suction Head	ft	512.7	165.6
Suction Specific Speed	- - -	39,323	19,106

TABLE 1 (cont.)

Turbine	Dimension	Fuel	Oxidizer
Shaft Power	hp	240,157	76,250
Gas Weight Flow Rate	lb/sec	183.9	124.7
Turbine Exhaust Thrust	lb	0	0
Total Gas Inlet Temperature	°F	1000	1000
Total Gas Inlet Pressure	psia	995	902
Pressure Ratio	- - -	22.1	20.0
Static Back Pressure	psia	45	45
Shaft Speed	rpm	8,227	2,866
Efficiency	%	55.6	26.6
Turbine Nozzle Area	in^2	37.61	28.14
Turbine Pitch Dia.	in	40	40

TABLE 1 (cont.)

Gas Generator	Dimension	Fuel or Single GG	Oxidiser
Chamber Pressure	psia	995	902
Chamber Temperature	°F	1000	1000
Total Propellant Flow Rate	lb/sec	183.9	124.8
Oxidiser Weight Flow Rate	lb/sec	81.7	55.5
Fuel Weight Flow Rate	lb/sec	102.2	69.3
Mixture Ratio ($\dot{w}_{OG}/\dot{w}_{FG}$)	- - -	0.8	0.8
Characteristic Velocity	ft/sec	6,547	6,547
Ratio of Specific Heats (k)	- - -	1.37	1.37
Gas Constant (R)	ft/°R	421.5	421.5
Pressure Ratio P_{cGG}/P_{TIT}	- - -	1.00	1.00

TABLE 1 (cont.)

THRUST						
MR						
PROPELLANT SYSTEM	OXID	FUEL	OXID	FUEL	OXID	FUEL
Chamber, Plenum	1000					
Chamber, Injector Face	1052					
ΔP Injector	132	80				
Injector Inlet	1184	1132				
ΔP Cooling Jacket	---	125				
Coolant Jacket Inlet	---	1257				
ΔP Discharge Line	15	15				
TCV Outlet	1199	1272				
ΔP TCV	80	36				
TCV Inlet	1279	1308				
ΔP P.U. Valve	---	---				
Pump Discharge	1279	1308				
TURBINE SYSTEM	Oxidizer Turbine		Fuel Turbine			
Turbine Inlet	902		995			
ΔP Orfice ΔP G G & Ducting						
G G Injector Face	902		995			
ΔP Injector	115	115	120	120		
Injector Inlet	1017	1017	1115	1115		
ΔP G G V	138	165	66	93		
GGV Inlet	1155	1182	1181	1208		
ΔP Orfice & Line	76	101	50	75		
GG Tap-off	1231	1283	1231	1283		

TABLE 2

PERFORMANCE AND OPERATING PARAMETERS, ENGINE PERFORMANCE AT STATIC CONDITION

Engine	Dimension	Sea Level Values
Ambient Pressure	psia	14.697
Acceleration a/g	— -	1.00
Instantaneous Specific Impulse (Engine)	sec	342.09
Engine Mixture Ratio	- - -	5.56
Total Engine Thrust (Excluding Turbine exhaust)	lb	5,828,188
TCA		
Thrust	lb	5,828,188
Thrust Chamber Pressure (Plenum Total)	psia	978.1
Thrust Chamber Pressure (Injector face)	psia	1029
Instantaneous Specific Impulse (TC)	lb/lb/sec	348.3
Thrust Coefficient	- - -	1.428
Characteristic Velocity	ft/ sec	7849
Effective Exhaust Velocity	ft/sec	11,206
Nozzle Exit Design Pressure	psia	5.142
Nozzle Pressure Ratio (P_{op}/P_o	- - -	190.2
Total Propellant Flow Rate	lb/sec	16,734
Mixture Ratio $\dot{w}_o/\dot{w}_f$	- - -	5.889
Fuel Flow Rate	lb/sec	2429
Oxidizer Flow Rate	lb/sec	14,305
Nozzle Area Ratio	- - -	20
Nozzle Throat Area	in^2	3968

TABLE 2 (cont.)

Turbo Pump	Dimension	Fuel	Oxidizer
Ambient Temperature	°F	60	60
Propellant Temperature			
Inlet	°F	-421	-294
Outlet	°F	-406	-294
Propellant Density			
Inlet	lb/ft^3	4.37	70.4
Outlet	lb/ft^3	4.44	70.4
Propellant Vapor Pressure	psia	20.8	19.4
Shaft Speed	rpm	8,138	2,858
Total Discharge Pressure	psia	1,280	1,243
Static Discharge Pressure	psia		
Total Suction Pressure	psia	35.88	34.60
Static Suction Pressure	psia		
Total Pressure Rise	psi	1,244.12	1,158.40
Total Head Rise	ft	40,348	2370
Propellant Flow Rate	lb/sec	2,598	14,438
Capacity	gpm	262,666	92,046
Specific Speed	- - -	1,465	2,552
Efficiency	%	81.94	83.71
Fluid Horsepower	hp	190,628	62,220
Shaft Horsepower	hp	232,657	74,330
Net Positive Suction Head	ft	489.1	133.3
Suction Specific Speed	- - -	40,104	22,102

TABLE 2 (cont.)

Turbine	Dimension	Fuel	Oxidizer
Shaft Power	hp	232,657	74,330
Gas Weight Flow Rate	lb/sec	180.6	122.6
Turbine Exhaust Thrust	lb	0	0
Total Gas Inlet Temperature	°F	981.4	986.4
Total Gas Inlet Pressure	psia	974.2	884.2
Pressure Ratio	- - -	21.65	19.65
Static Back Pressure	psia	45	45
Shaft Speed	rpm	8,138	2,858
Efficiency	%	55.48	26.62
Turbine Nozzle Area	in^2	37.61	28.14
Turbine Pitch Dia.	in	40	40

TABLE 2 (cont.)

Gas Generator	Dimension	Fuel or Single GG	Oxidizer
Chamber Pressure	psia	974.2	884.2
Chamber Temperature	°F	981.4	986.4
Total Propellant Flow Rate	lb/sec	180.6	122.6
Oxidizer Weight Flow Rate	lb/sec	79.63	54.16
Fuel Weight Flow Rate	lb/sec	101.0	68.43
Mixture Ratio ($\dot{w}_{OG}/\dot{w}_{FG}$)	- - -	.7884	.7915
Characteristic Velocity	ft/sec	6,526	6,531
Ratio of Specific Heats (k)	- - -	1.370	1.370
Gas Constant (R)	ft/°R	424.3	423.5
Pressure Ratio P_{cGG}/P_{TIT}	- - -	1.00	1.00

TABLE 2 (cont.)

[illegible]						
[illegible]						
PROPELLANT SYSTEM	OXID.	FUEL	OXID.	FUEL	OXID.	FUEL
Chamber, Plenum	978.1					
Chamber, Injector Face	1029					
ΔP Injector	125	78				
Injector Inlet	1154	1107				
ΔP Cooling Jacket	---	123				
Coolant Jacket Inlet	---	1230				
ΔP Discharge Line	14	15				
TCV Outlet	1168	1245				
ΔP TCV	75	35				
TCV Inlet	1243	1280				
ΔP P.U. Valve	---	---				
Pump Discharge	1243	1280				
TURBINE SYSTEM	Oxidizer Turbine		Fuel Turbine			
Turbine Inlet	884		974			
ΔP Orfice ΔP G G & Ducting	---		---			
G G Injector Face	884		974			
ΔP Injector	110	112	114	117		
Injector Inlet	994	996	1088	1091		
ΔP G G V	132	161	63	92		
GGV Inlet	1126	1157	1151	1183		
ΔP Orfice & Line	72	99	47	73		
GG Tap-off	1198	1256	1198	1256		

TABLE 3

EFFECT OF ENGINE TOLERANCES UPON ENGINE PARAMETERS

Parameter	Tolerance	Engine Thrust Variation, Percent	Engine Mixture Ratio Variation, Percent	Engine ISP Variation, Percent	Chamber Pressure Variation, Percent
Oxidizer Tank Pressurization System Pressure	+ 2 Psi	+ 0.2094	+ 0.0309	+ 0.0318	+ 0.1727
	- 2 Psi	- 0.1687	- 0.2159	- 0.0109	- 0.1173
Fuel Tank Pressurization System Pressure	+ 2 Psi	+ 0.0198	- 0.1374	+ 0.0237	+ 0.0822
	- 2 Psi	+ 0.0789	+ 0.0106	+ 0.0024	+ 0.0154
Oxidizer Suction Temperature	+ 2° F	- 0.3295	- 0.3728	- 0.0250	- 0.2341
Fuel Suction Temperature	+ 2° F	- 0.5226	+ 1.2697	- .1954	- .5863
Oxidizer Pump Head	+ 1%	+ 0.9205	+ 0.5860	+ 0.1029	+ 0.7079
	- 1%	- 0.8779	- 0.6773	- 0.0930	- 0.6614
Fuel Pump Head	+ 1%	+ 0.2587	- 0.4959	+ 0.0807	+ 0.2750
	- 1%	- 0.2434	+ 0.5062	- 0.0804	- 0.2625
Hydraulic Resistance	+ 1%	- 0.2720	- 0.1344	- 0.0337	- 0.2136
	- 1%	+ 0.3757	- 0.0055	+ 0.0612	+ 0.3168
RMS TOTAL = $\sqrt{\leq N^2}$		+ 1.0515	+ 1.4876	+ 0.1498	+ 0.8450
		- 1.1463	- 1.0269	- 0.2336	- .9462

TABLE 4

EFFECT OF ENGINE TOLERANCES UPON ENGINE PARAMETERS AT CONSTANT MIXTURE RATIO

Parameter	Tolerances	Thrust Variation Percent	Chamber Pressure Variation Percent	Engine ISP Variation Percent
Oxidizer Tank Pressurization System Pressure	+ 2 Psi	+ .023611	+ .020148	+ .004579
	- 2 Psi	+ .002318	+ .002226	- .000571
Fuel Tank Pressurization System Pressure	+ 2 Psi	+ .159988	+ .134942	+ .025705
	- 2 Psi	- .157780	- .132534	- .025263
Oxidizer Suction Temperature	+ 2° F	+ .085970	+ .072639	+ .011521
Fuel Suction Temperature	+ 2° F	- 2.428	- 2.043	- .398
Oxidizer Pump Head	+ 1%	+ .033144	+ .02817	+ .009665
	- 1%	- .021970	- .018218	+ .007717
Fuel Pump Head	+ 1%	+ 1.016826	+ .856121	+ .161713
	- 1%	- .998361	- .840117	- .161468
Hydraulic Resistances	+ 1%	- .224202	- .188445	- .036539
	- 1%	+ .221828	+ .186991	+ .036175
RMS TOTAL $=\sqrt{\Sigma N^2}$		+ 1.057256	+ .890280	+ .168368
		- 2.6396	- 2.2210	- .4318

TABLE 5

EFFECT OF ENGINE TOLERANCES UPON ENGINE PARAMETERS, VOLUME MEASURING PROPELLANT UTILIZATION SYSTEM

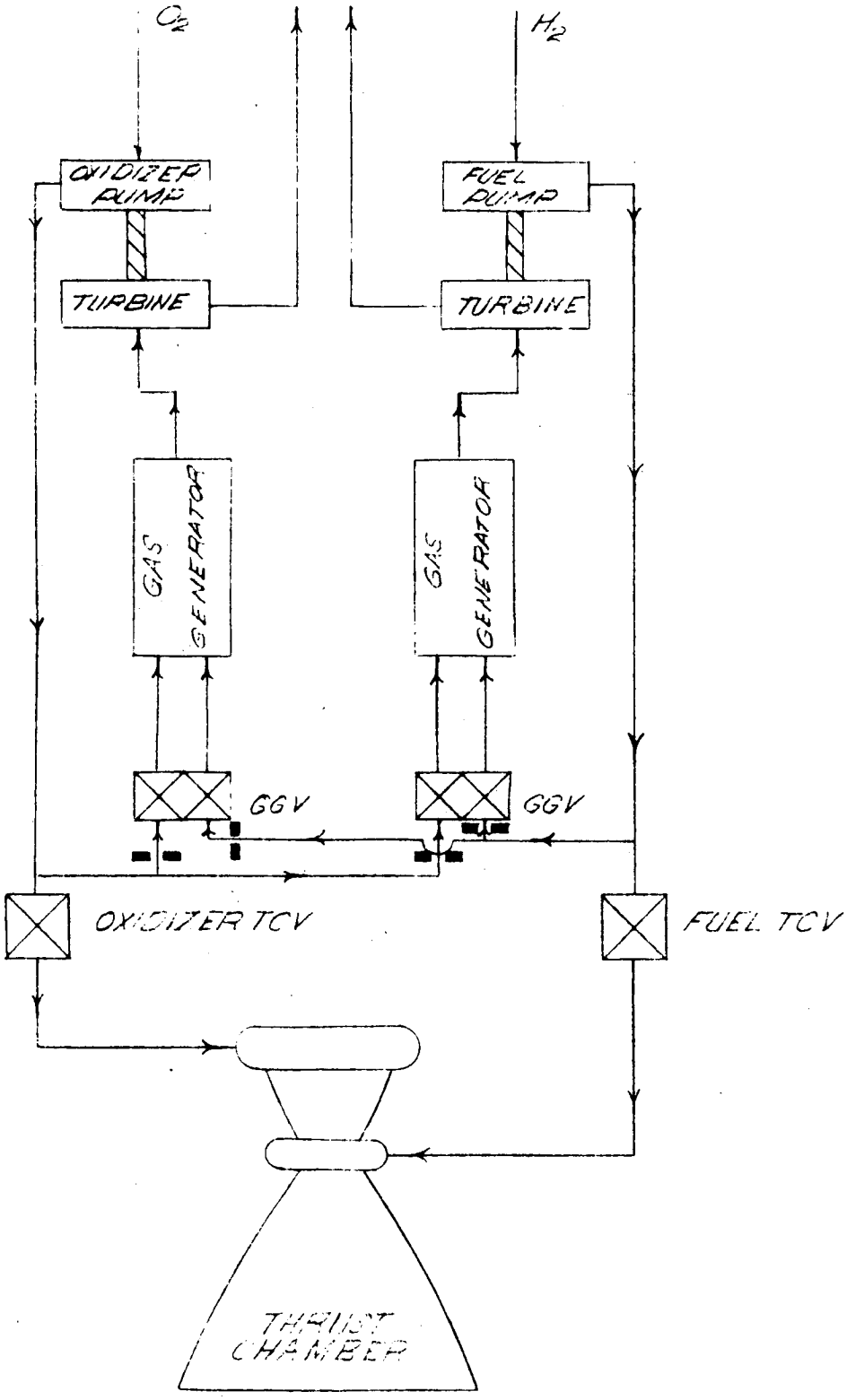
Parameter	Tolerances	Thrust Variation Percent	Chamber Pressure Variation Percent	Engine ISP Variation Percent
Oxidizer Suction Pressure	+ 2 Psi - 2 Psi	+ .023611 + .002318	+ .020148 + .002226	+ .004579 - .000571
Fuel Suction Pressure	+ 2 Psi - 2 Psi	+ .159988 - .157780	+ .134942 - .132534	+ .025705 - .025263
Oxidizer Suction Temperature	+ 2° F	+ .085970	+ .072639	+ .011521
Fuel Suction Temperature	+ 2° F	+ .197000	- .036121	- .107800
Oxidizer Pump Head	+ 1% - 1%	+ .033144 - .021970	+ .02817 - .018218	+ .009665 + .007717
Fuel Pump Head	+ 1% - 1%	+ 1.016826 - .998361	+ .856121 - .840117	+ .161713 - .161468
Hydraulic Resistances	+ 1% - 1%	- .224202 + .221828	- .188445 + .186991	- .036539 + .036175
RMS TOTAL = $\sqrt{\varepsilon N^2}$		+ 1.074942 - 1.035552	+ .890280 - .8720722	+ .168368 - .19909779

Table 5

TABLE 6

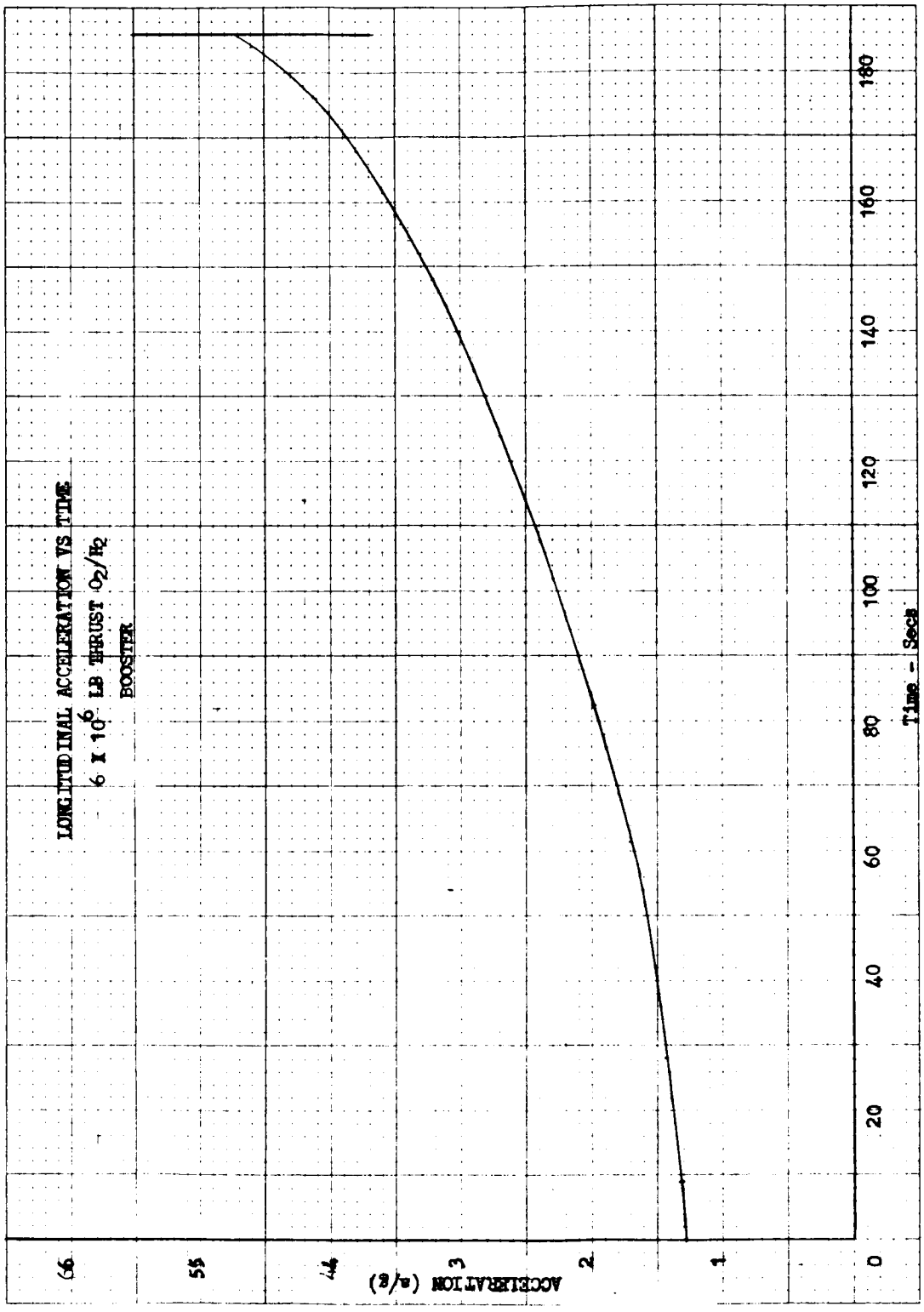
EFFECT OF ENGINE TOLERANCES UPON ENGINE PARAMETERS AT CONSTANT THRUST CONTROL

Parameter	Tolerances	Engine Mixture Ratio Variation Percent	Engine ISP Variation Percent
Oxidizer Tank Pressurization System Pressure	+ 2 Psi	- .008304	+ .001812
	- 2 Psi	- .013957	+ .000023
Fuel Tank Pressurization System Pressure	+ 2 Psi	- .130207	+ .82441
	- 2 Psi	+ .114660	- .006613
Oxidizer Suction Temperature	+ 2° F	- .072436	+ .002080
	+ 2° F	+ 1.639	- .1441
Oxidizer Pump Head	+ 1%	- .021201	+ .006703
	- 1%	+ .004593	- .005046
Fuel Pump Head	+ 1%	- .673828	+ .054274
	- 1%	+ .660577	- .055428
Hydraulic Resistances	+ 1%	+ .149995	- .012663
	- 1%	- .128794	+ .011728
RMS TOTAL = $\sqrt{\Sigma N^2}$		+ 1.7772 - .702528	+ .826310 - .155134



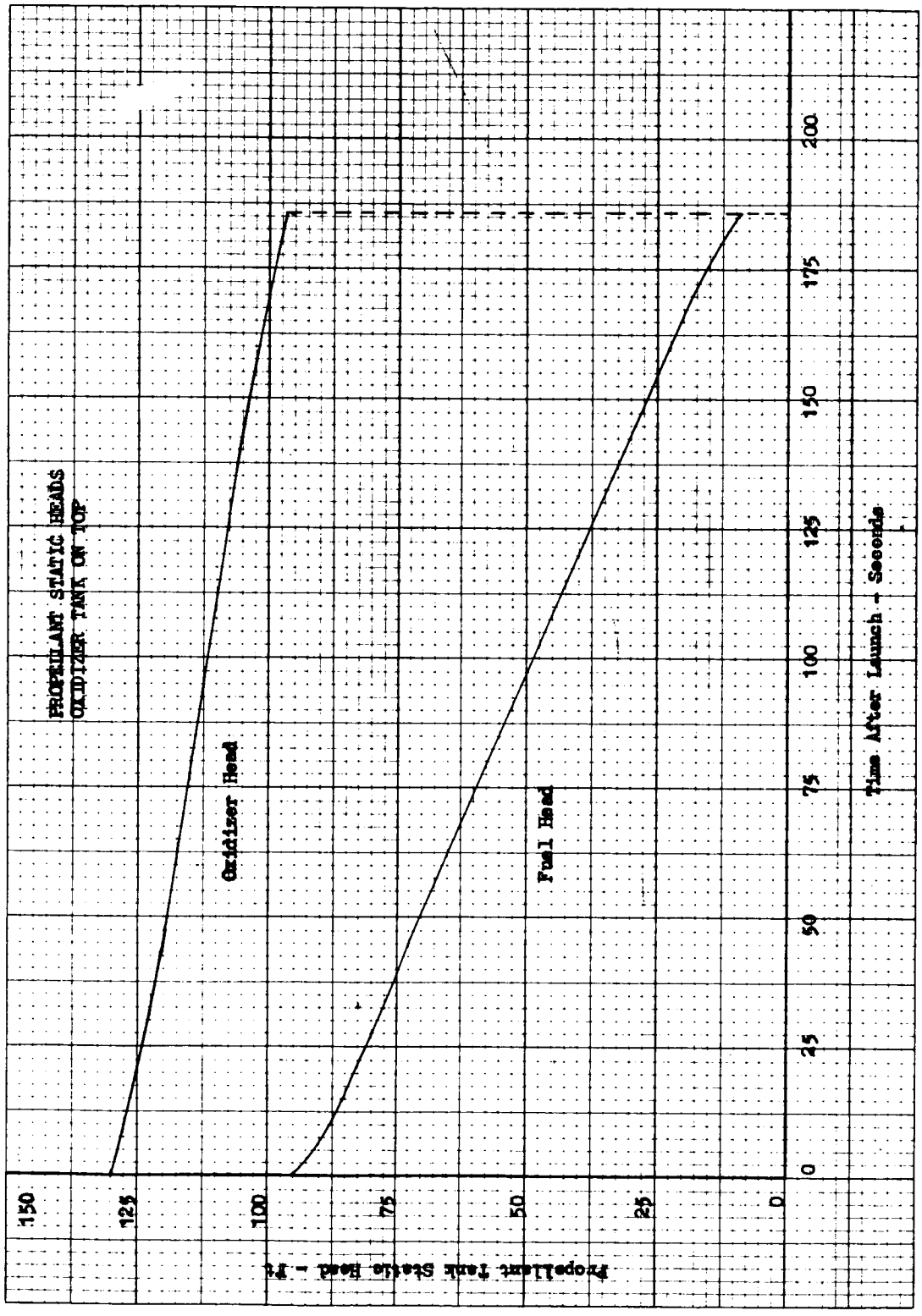
Engine System Schematic

Figure 1



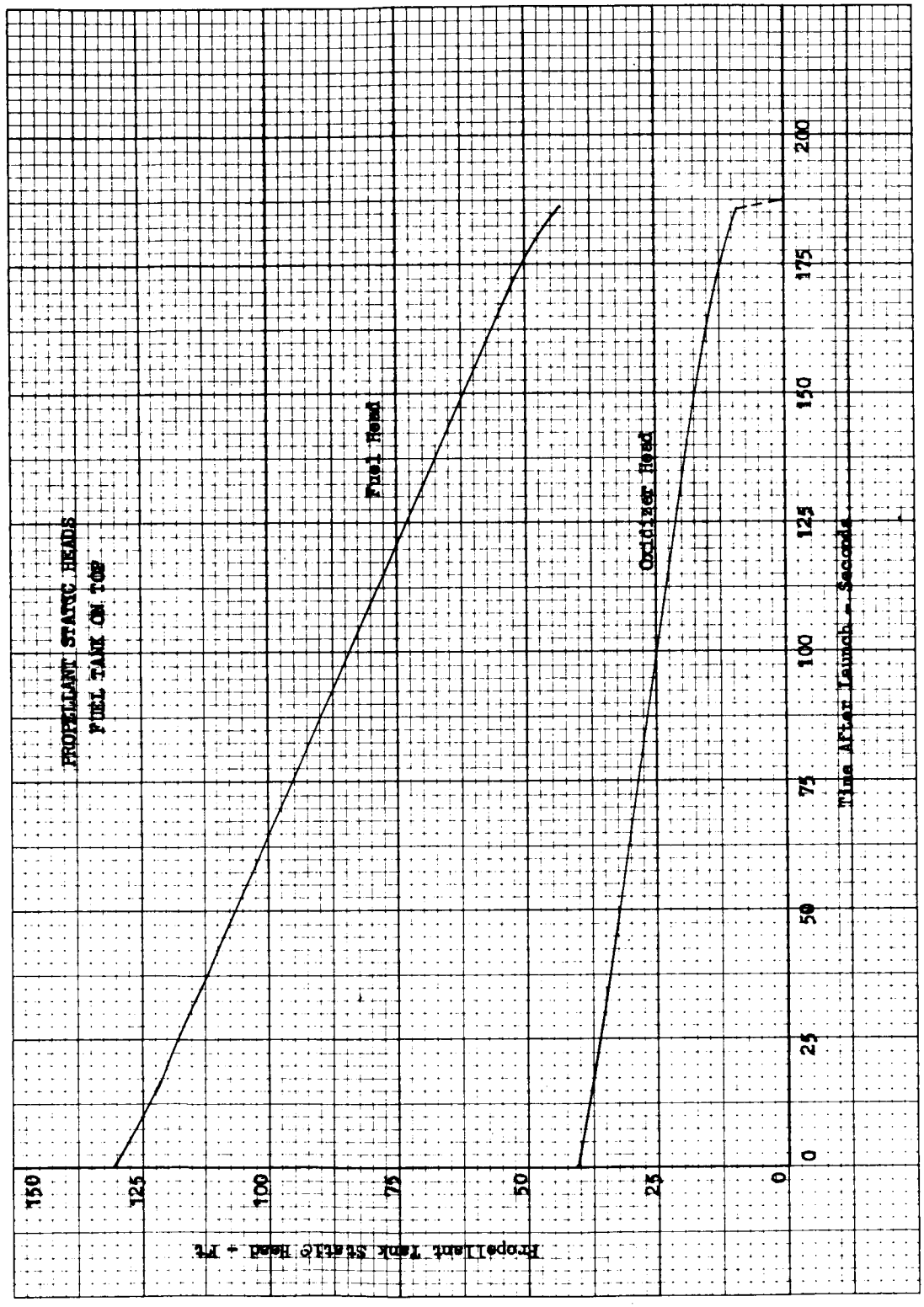
Longitudinal Acceleration vs Time, 6 x 10^6 lb Thrust O_2/H_2 Booster

Figure 2



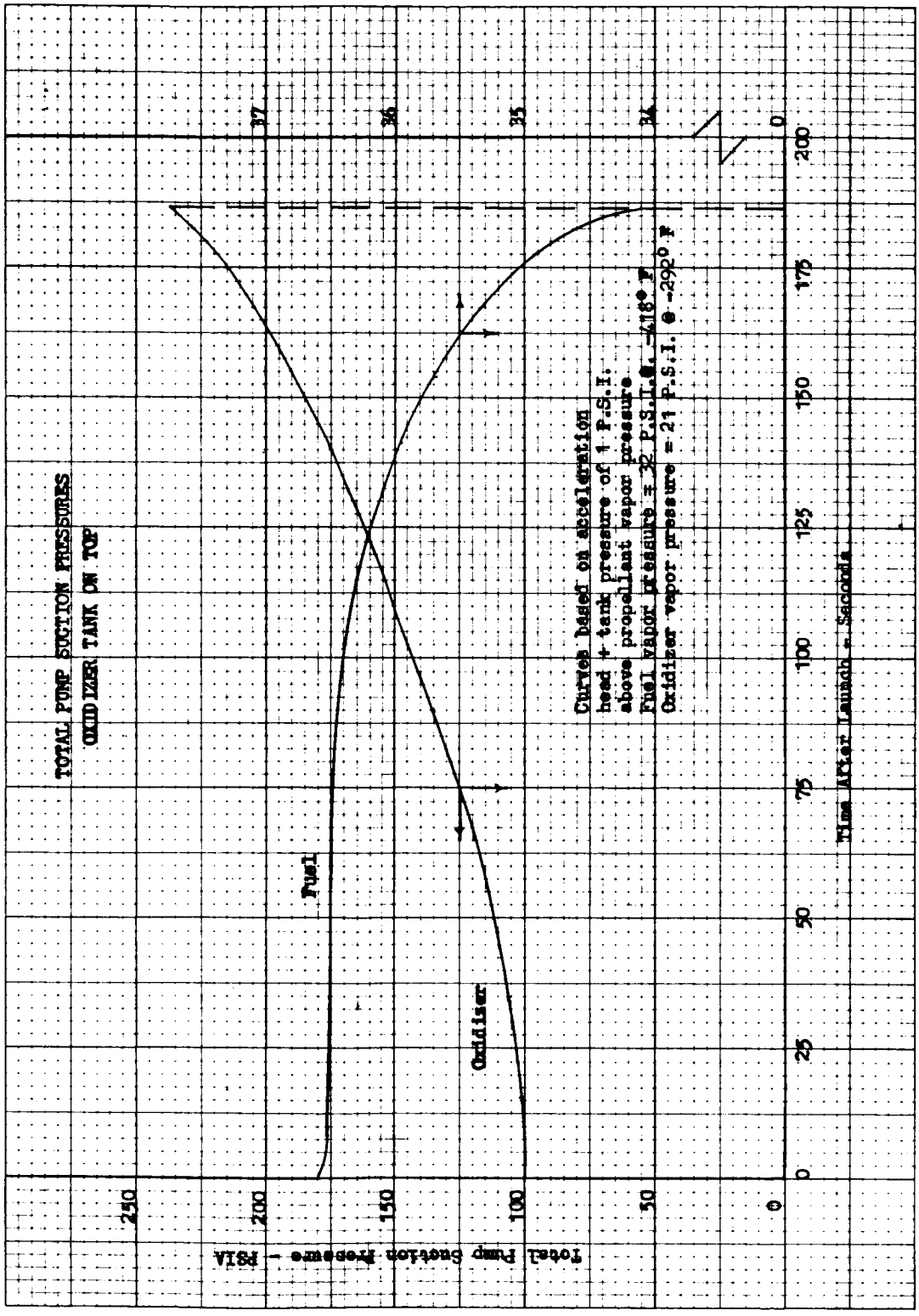
Propellant Static Heads, Oxidizer Tank on Top

Figure 3



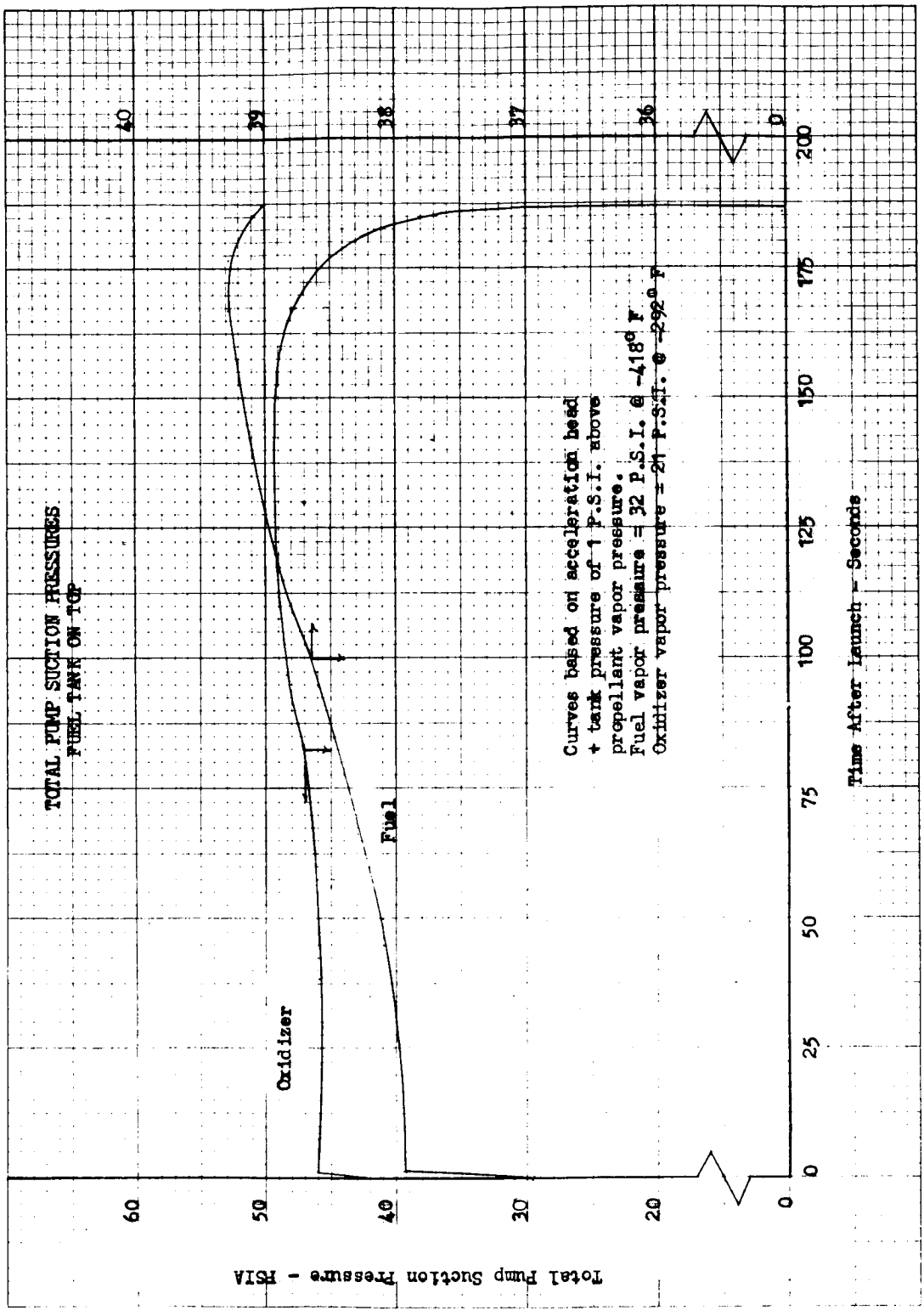
Propellant Static Heads, Fuel Tank on Top

Figure 4



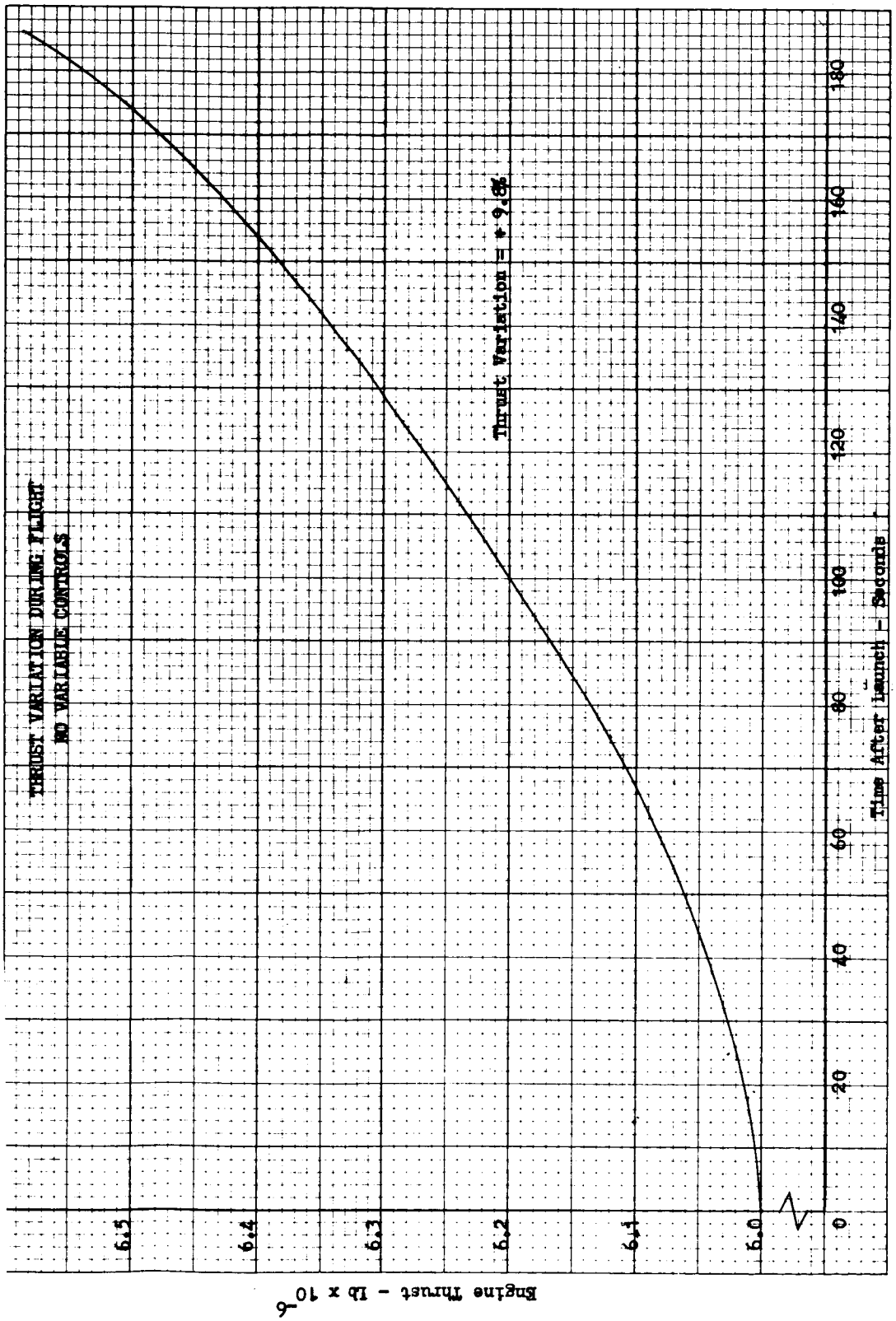
Total Pump Suction Pressures, Oxidizer Tank on Top

Figure 5



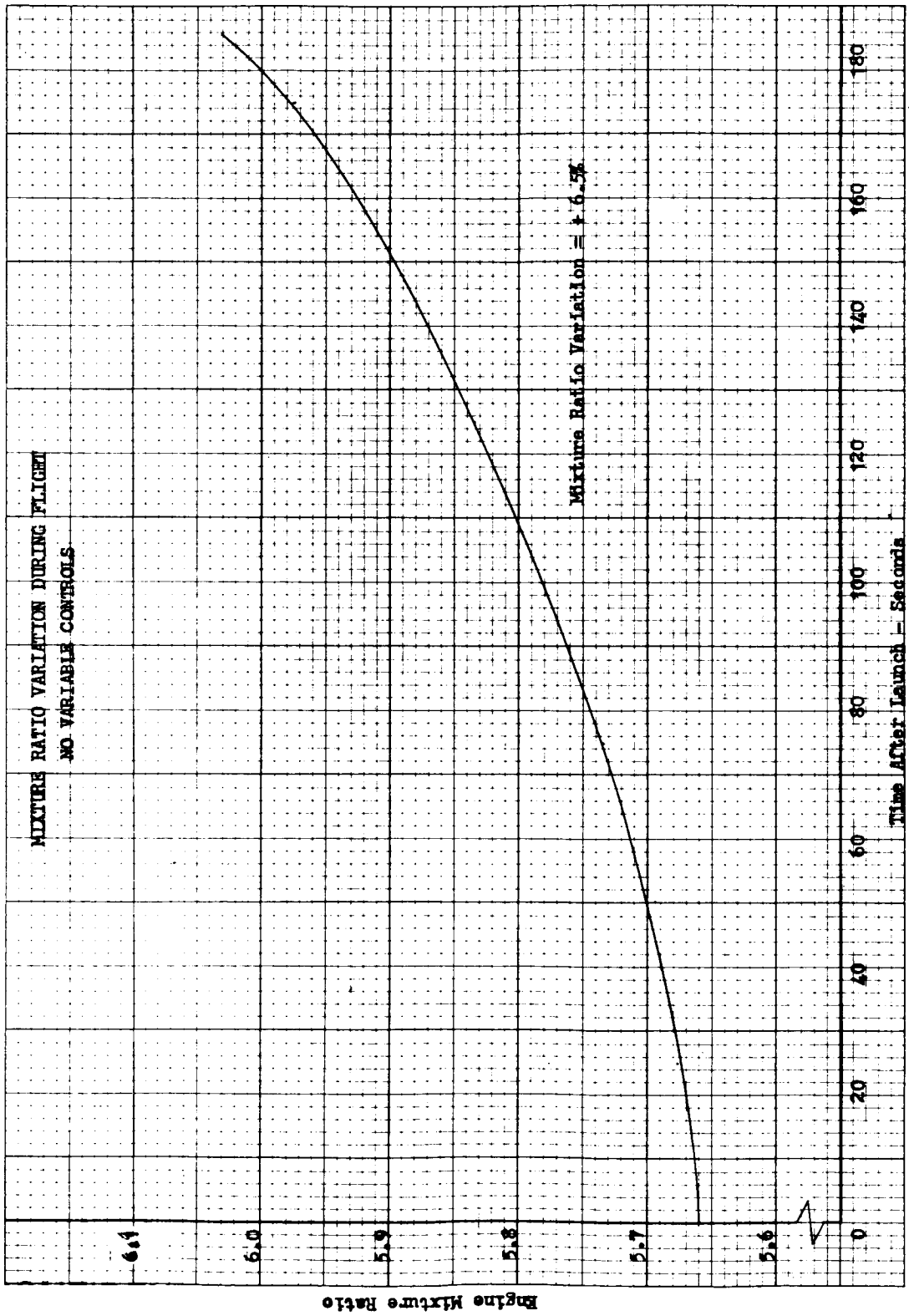
Total Pump Suction Pressures, Fuel Tank on Top

Figure 6



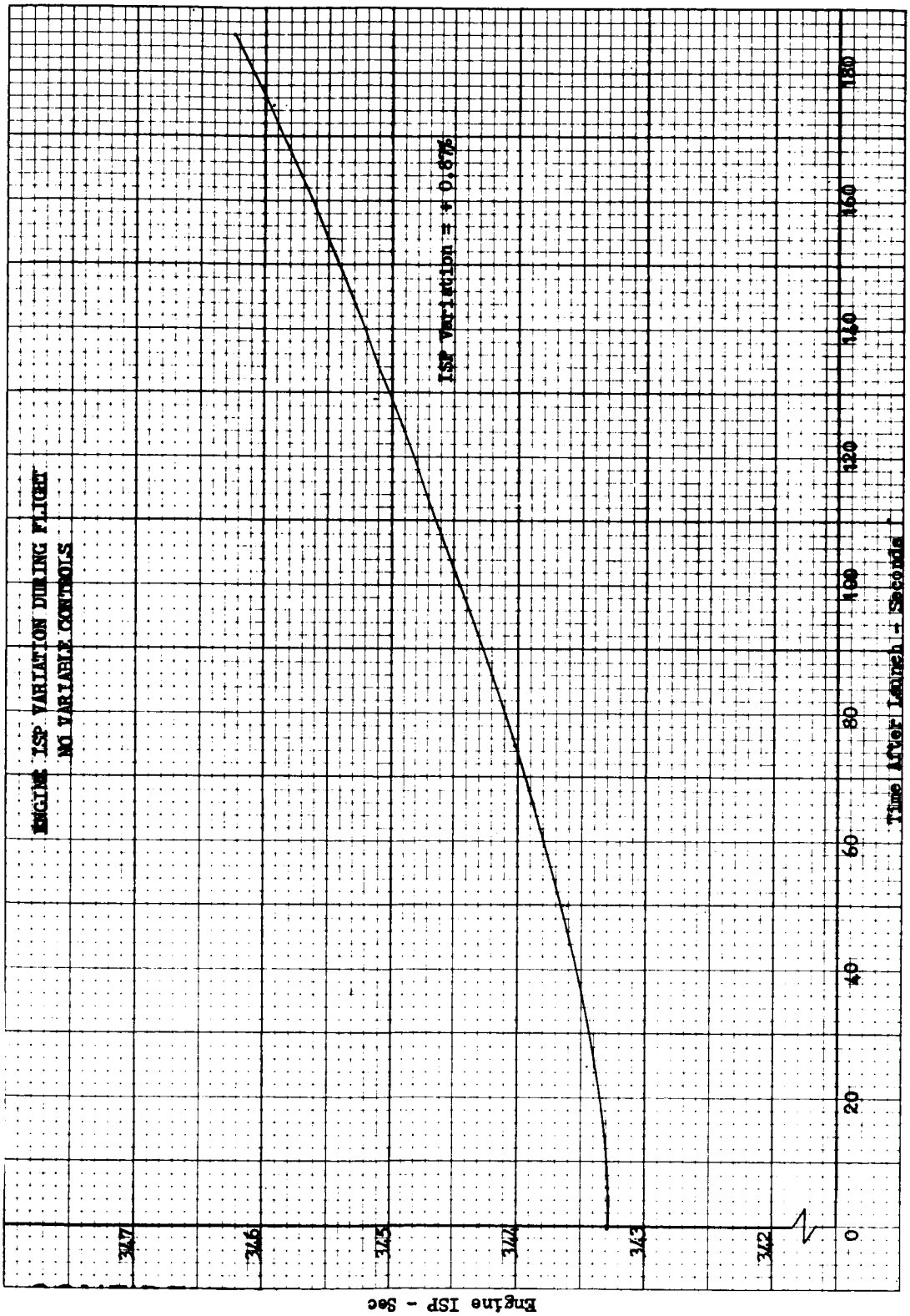
Thrust Variation During Flight, No Variable Controls

Figure 7



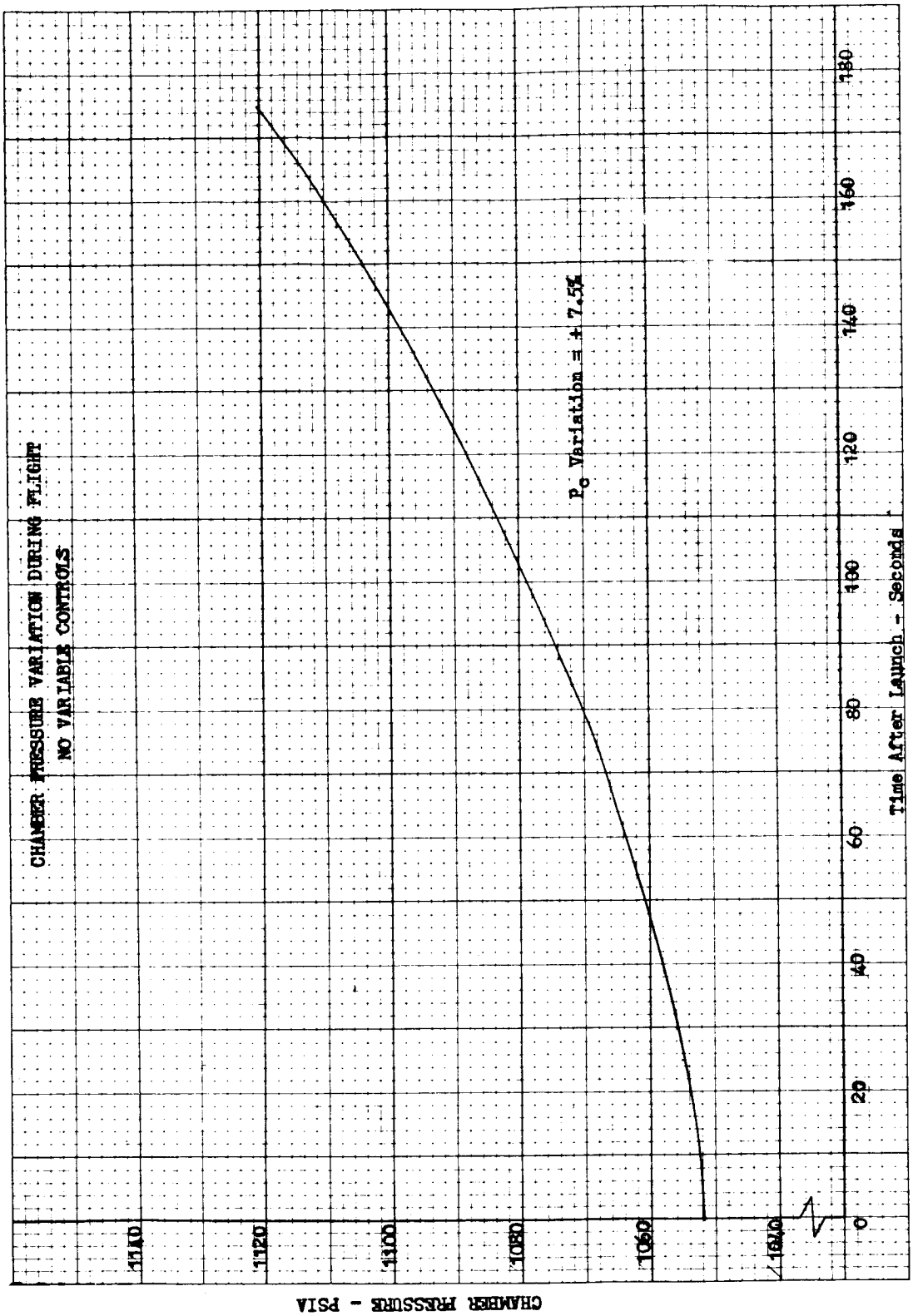
Mixture Ratio Variation During Flight, No Variable Controls

Figure 8



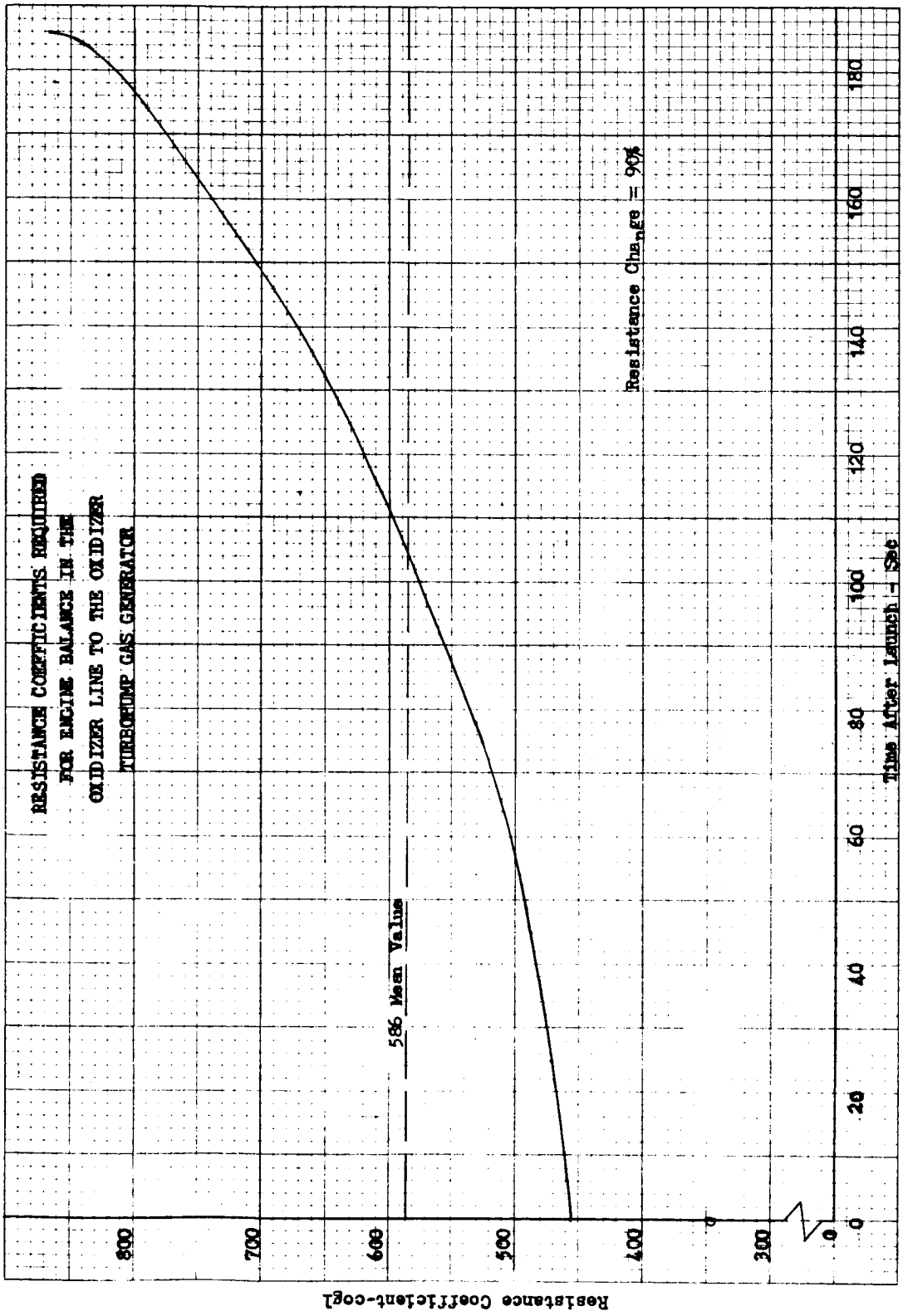
Engine I_{sp} Variation During Flight, No Variable Controls

Figure 9



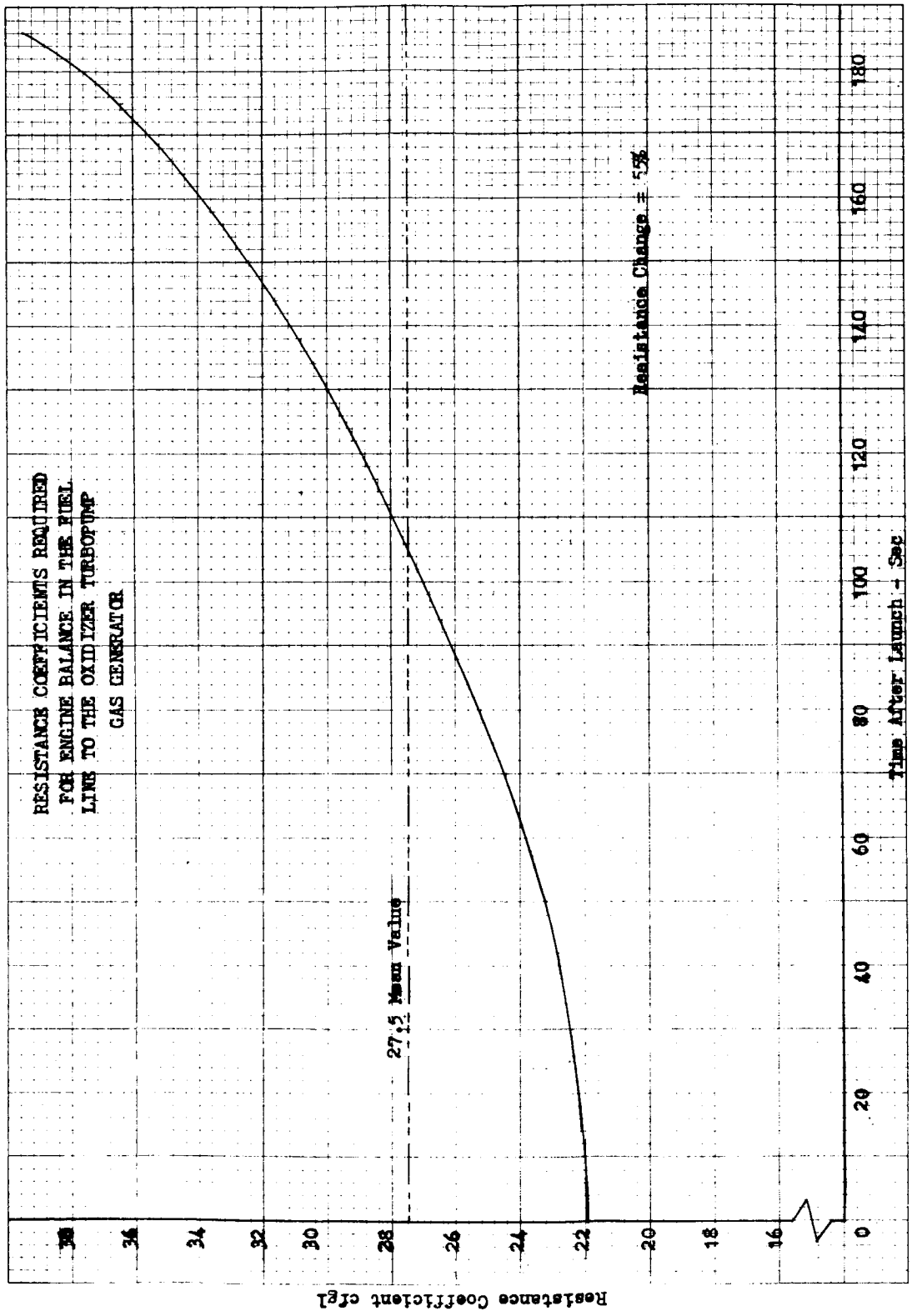
Chamber Pressure Variation During Flight, No Variable Controls

Figure 10



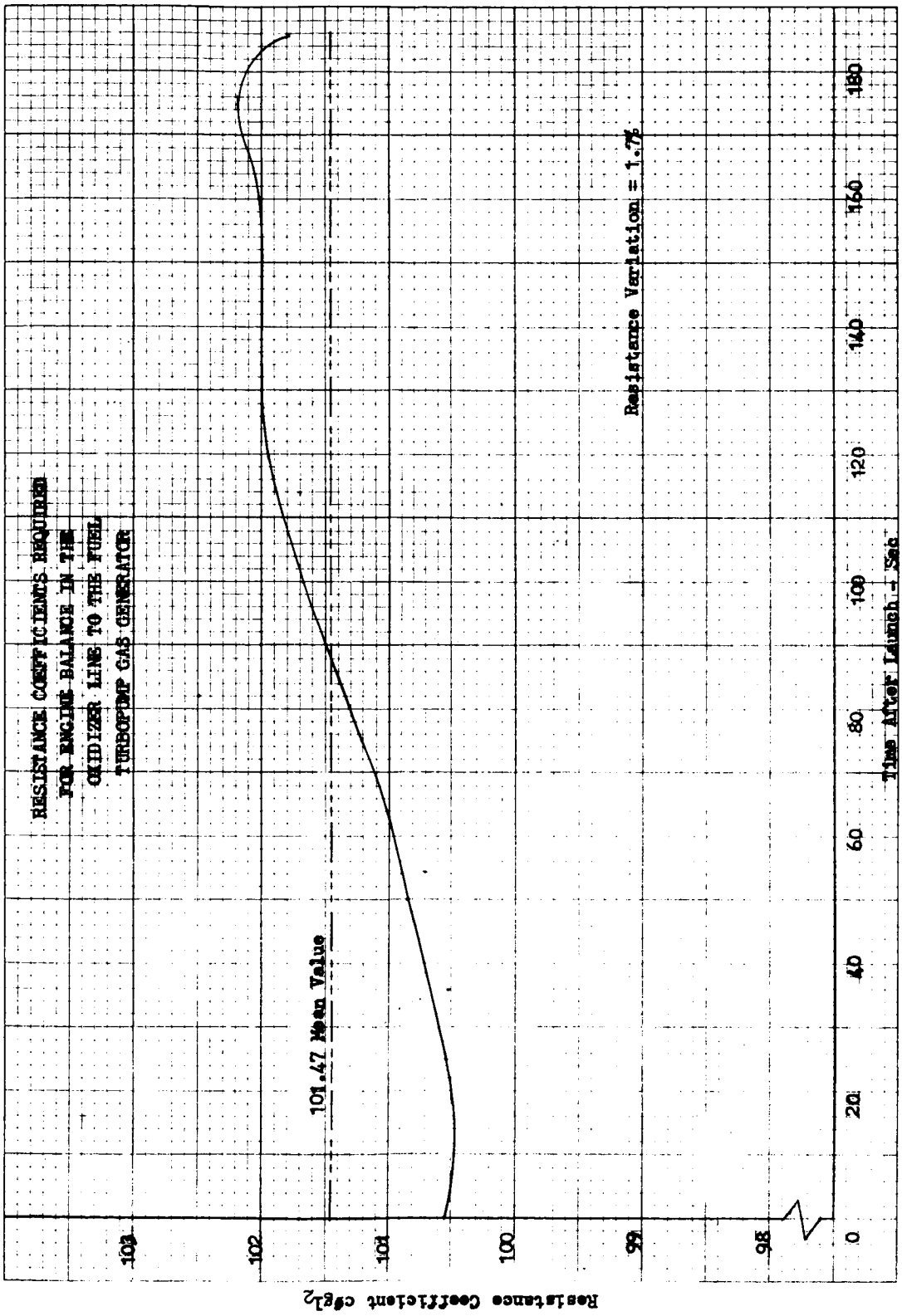
Resistance Coefficients Required for Engine Balance in the Oxidizer Line to the Oxidizer Turbopump Gas Generator

Figure 11



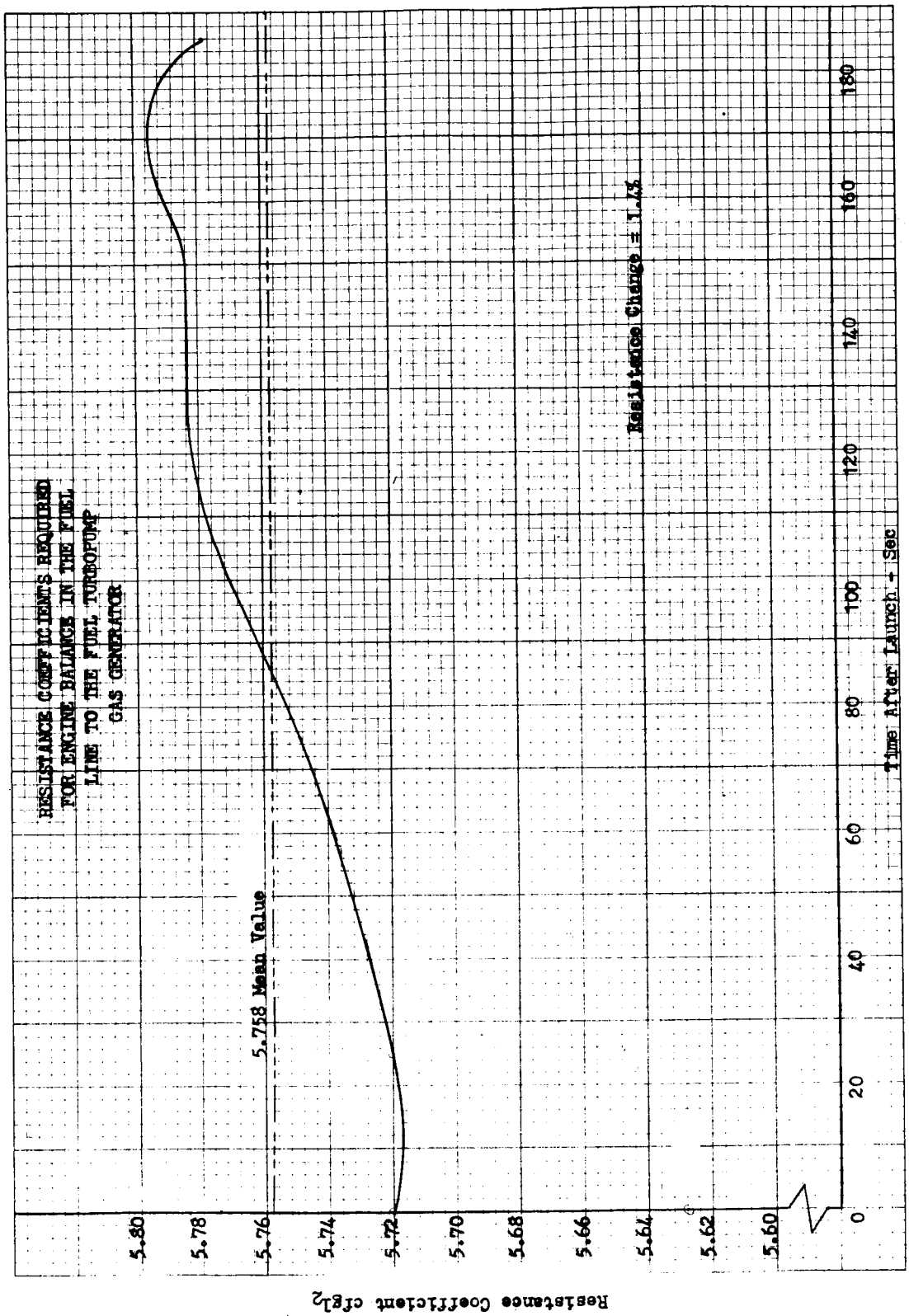
Resistance Coefficients Required for Engine Balance in the Fuel Line to the Oxidizer Turbopump Gas Generator

Figure 12



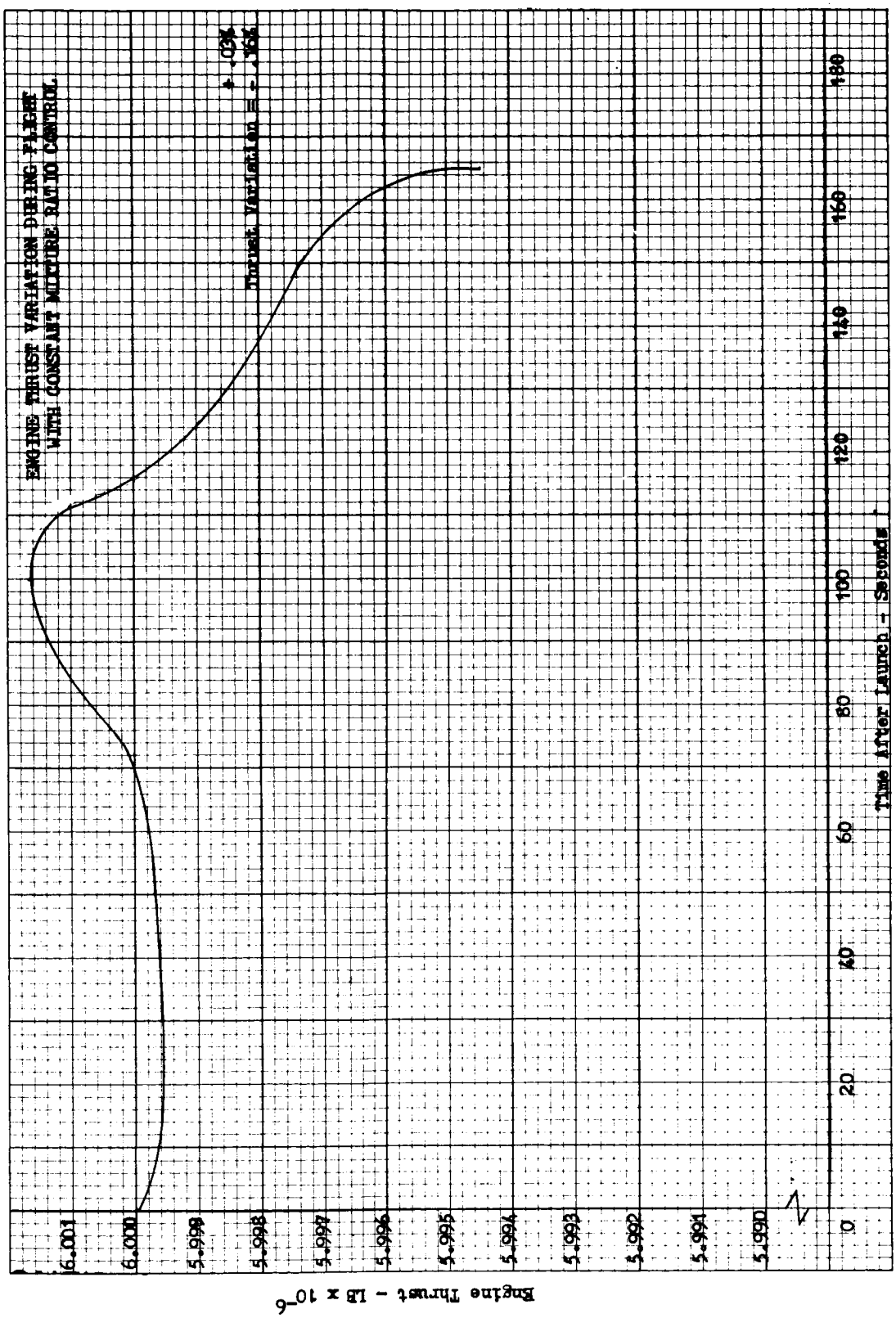
Resistance Coefficients Required for Engine Balance in the Oxidizer Line to the Fuel Turbopump Gas Generator

Figure 13



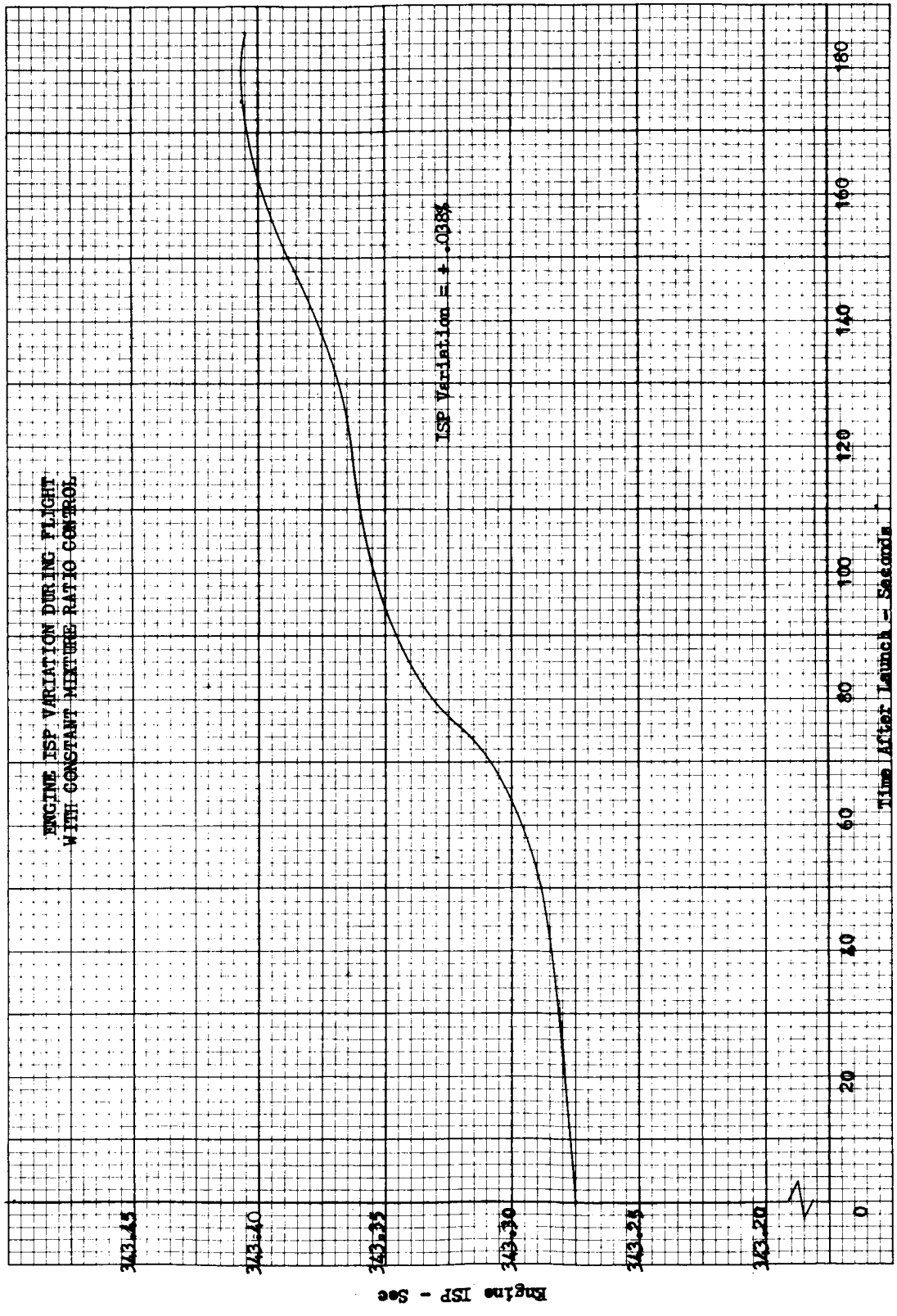
Resistance Coefficients Required for Engine Balance in the Fuel Line to the Fuel Turbopump Gas Generator

Figure 14



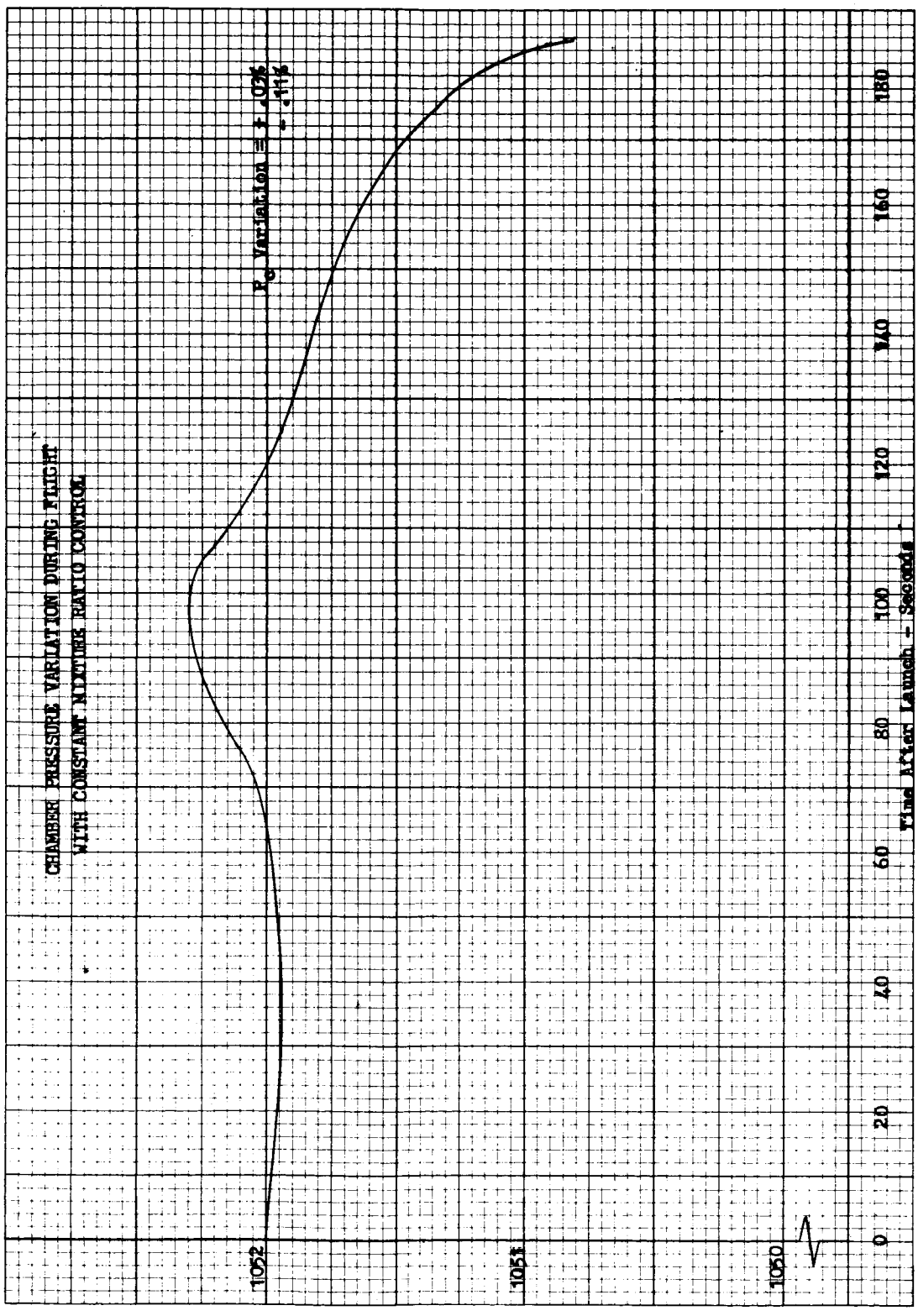
Engine Thrust Variation During Flight with Constant Mixture Ratio Control

Figure 15



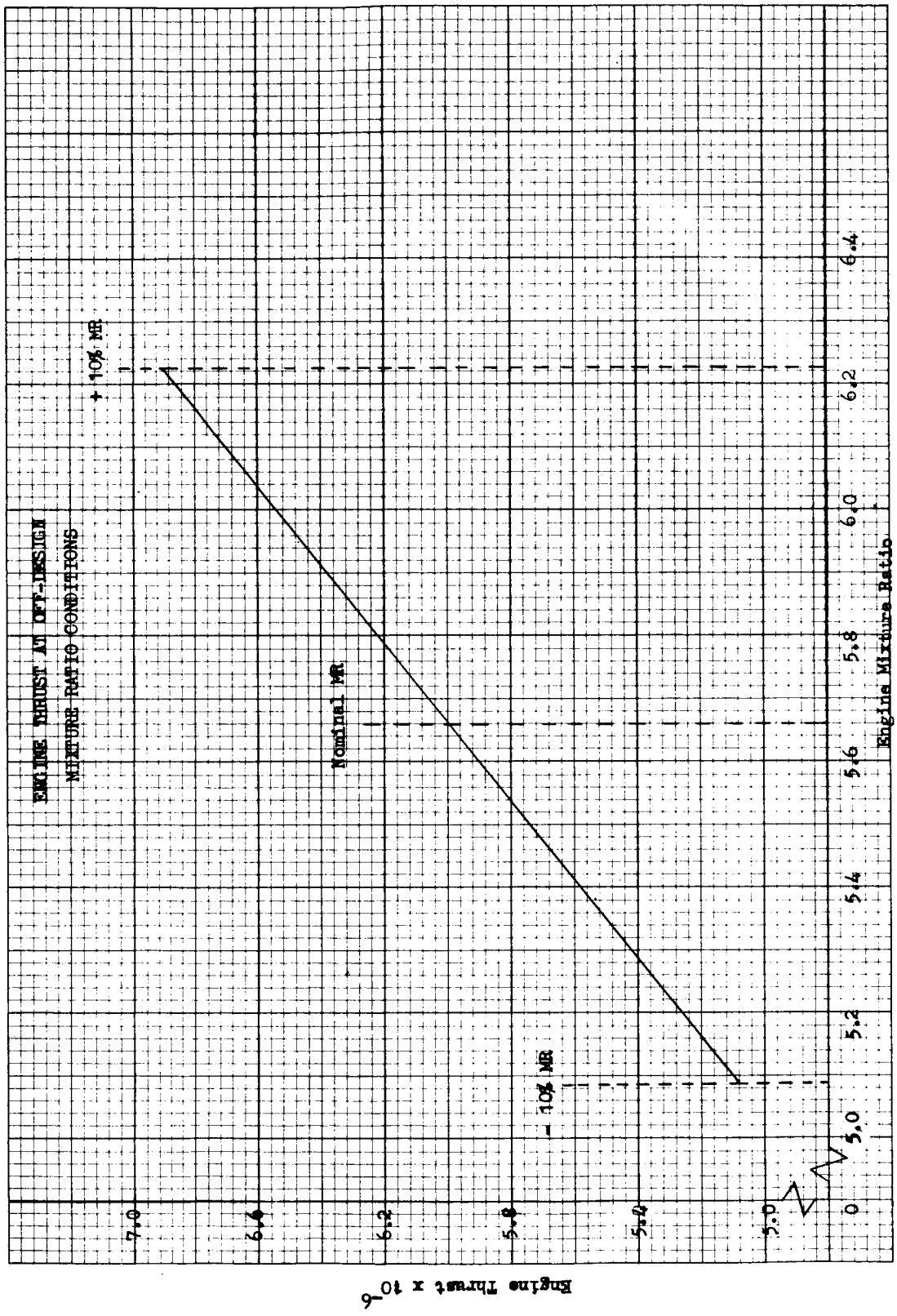
Engine I_{sp} Variation During Flight with Constant Mixture Ratio Control

Figure 16



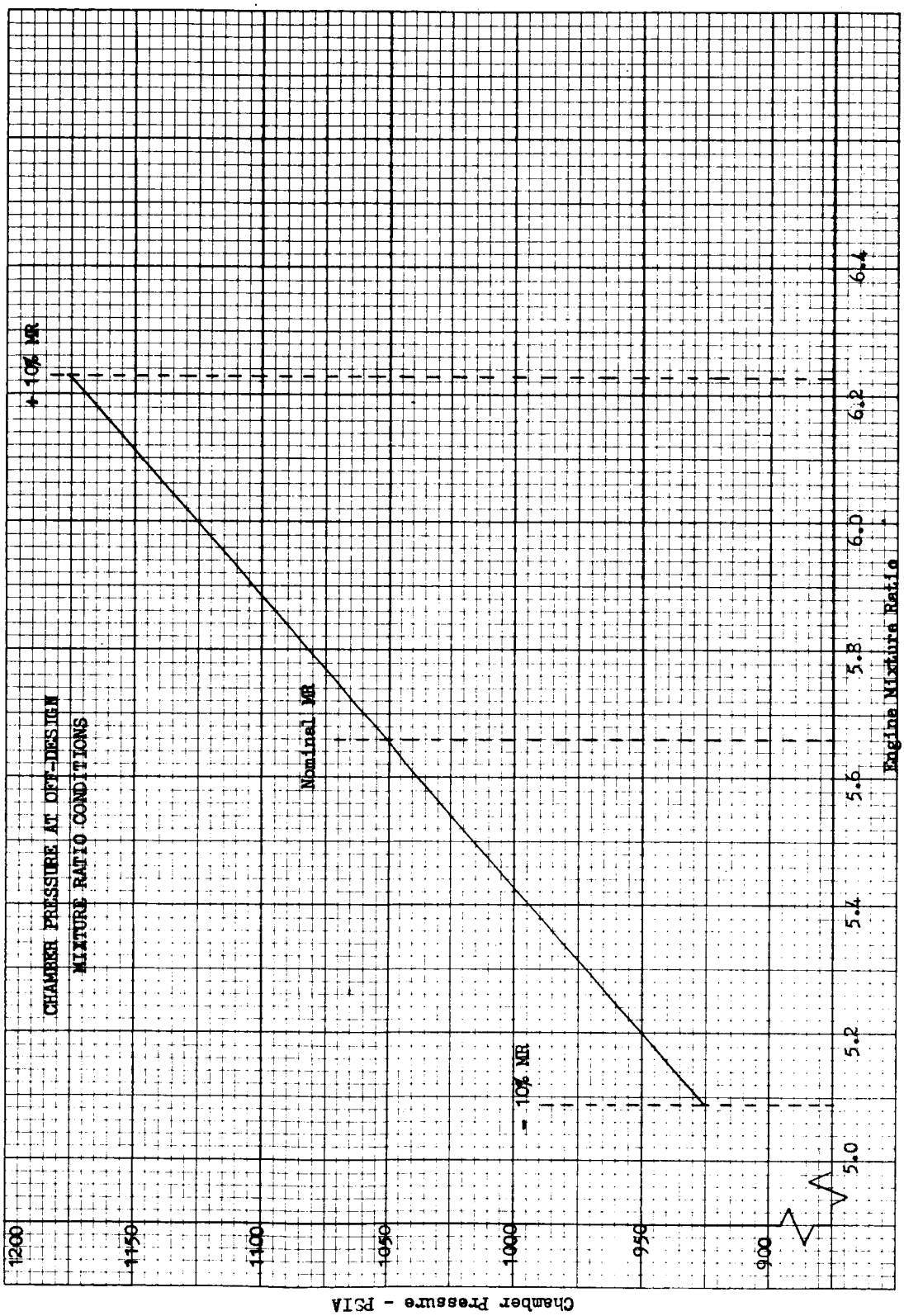
Chamber Pressure Variation During Flight with Constant Mixture Ratio Control

Figure 17



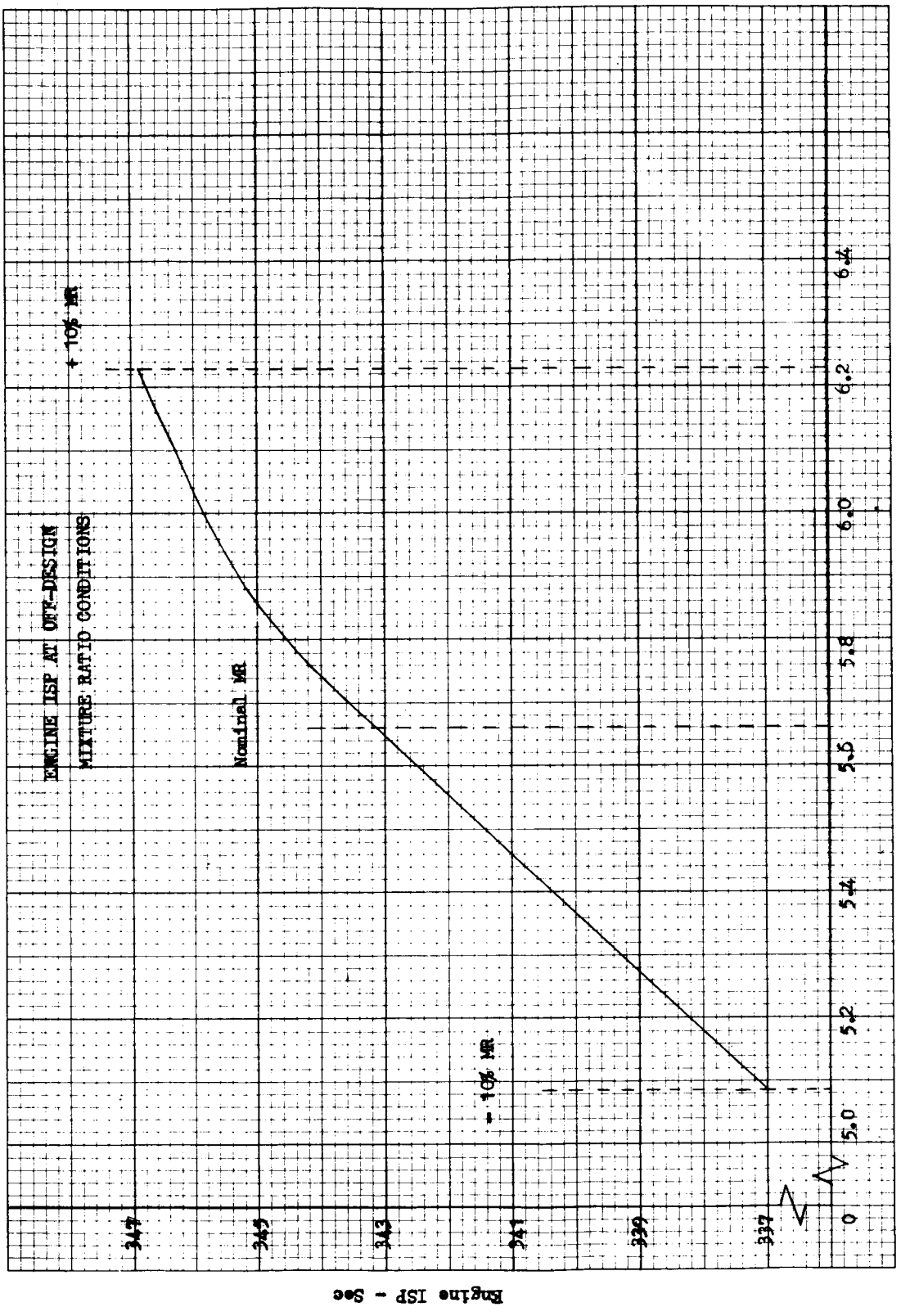
Engine Thrust at Off-Design Mixture Ratio Conditions

Figure 18



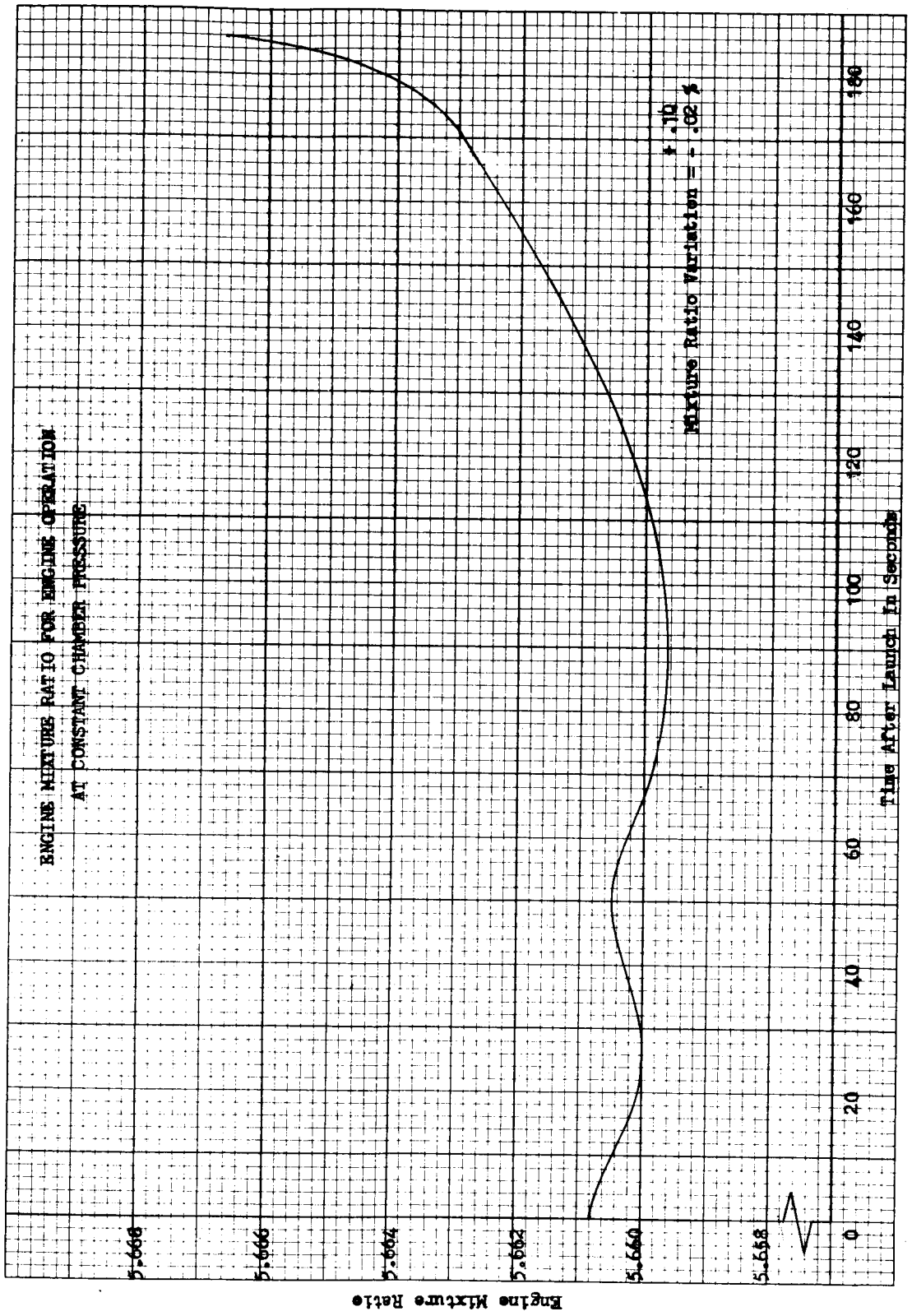
Chamber Pressure at Off-Design Mixture Ratio Conditions

Figure 19



Engine I_{sp} at Off-Design Mixture Ratio Conditions

Figure 20



Engine Mixture Ratio for Engine Operation at Constant Chamber Pressure

Figure 21

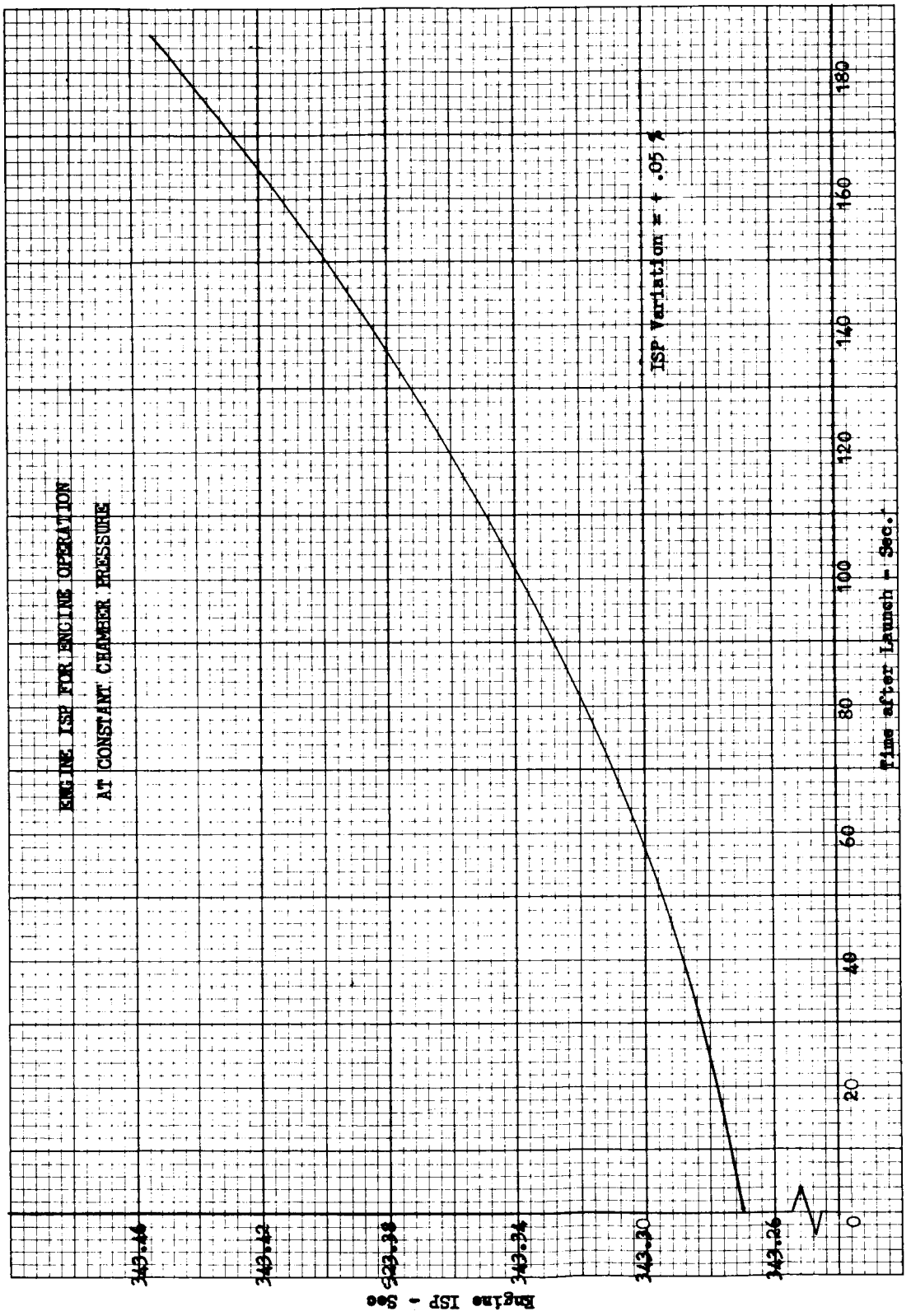


Figure 22

Engine Isp for Engine Operation at Constant Chamber Pressure

APPENDIX H

STUDY OF A LARGE PRESSURE-FED BOOSTER STAGE

TABLE OF CONTENTS

		PAGE NO.
I.	OBJECTIVE	I-1
II.	SUMMARY	II-1
III.	PHILOSOPHY AND SCOPE	III-1
IV.	BOOSTER SPECIFICATIONS	IV-1
	A. PARAMETER SELECTION	IV-1
	B. VEHICLE ANALYSIS	IV-4
V.	BOOSTER DESIGN STUDIES	V-1
	A. TANKAGE STUDY	V-1
	B. THRUST-CHAMBER ASSEMBLY	V-5
	C. PRESSURIZATION SYSTEM STUDIES	V-12
	D. VECTOR AND ROLL CONTROL	V-45
	E. PROPELLANT TANKAGE LOADING SYSTEM	V-48
	F. FACILITIES AND GROUND SUPPORT EQUIPMENT	V-51
VI.	INTEGRATION OF BOOSTER SUBSYSTEMS	VI-1
	A. SUBSYSTEM CONFIGURATION SELECTION	VI-1
	B. BOOSTER TRANSIENT OPERATION	VI-3
	C. PROPELLANT UTILIZATION CONSIDERATIONS	VI-5
	D. BOOSTER PRESSURE SCHEDULE	VI-6
	E. WEIGHT SUMMARY	VI-6
	F. FINAL TRAJECTORY COMPUTATION AND BOOSTER PERFORMANCE	VI-6

TABLE LIST

	TABLE NO.
300-nm Orbit Performance for a Pressure-Fed Booster With Pump-Fed Second Stage (O_2/H_2 Propellants)	IV-1
Pressurization Systems Weight Comparison	V-1
Control Moment Available by Chamber Pressure Modulation	V-2
Booster Pressure Schedule--Main Propellant System	VI-1
Booster Pressure Schedule--Pressurization System	VI-2
System Weight Summary--6,700K Pressure-Fed Engine	VI-3
300-nm Orbit Performance for a Pressure-Fed Booster With Pump-Fed Second Stage (O_2/H_2 Propellants)	VI-4

FIGURE LIST

	FIGURE NO.
Pressure-Fed Booster Configuration	II-1
Schematic, Pressure-Fed Booster Configuration	II-2
Optimum Staging Investigation for a 300-nm-Orbit Mission	IV-1
Booster Development Cost Per Pound of Payload Versus Booster Effective Ideal Velocity Increment	IV-2
Booster Production Cost Per Pound of Payload Versus Booster Effective Ideal Velocity Increment	IV-3
Pressure-Fed Booster Cost--Dollars Per Pound of Payload Versus Booster Effective Ideal Velocity Increment	IV-4
Acceleration Versus Time, 6×10^6-lb-Thrust Booster	IV-5
Conventional Pressure-Fed Vehicle, 6.7×10^6-lb Booster Thrust	IV-6
Vehicle Response at Assumed Kick-Angle Application--Standard Pressure-Fed Vehicle, 6.7×10^6-lb Thrust	IV-7
Thrust-Vector Angle Necessary to Accomplish Trajectory Kick--Standard Pressure-Fed Vehicle, 6.7×10^6-lb Thrust	IV-8
Standard Tankage Configuration	V-1
Alternative Tankage Configuration--No. 1	V-2
Alternative Tankage Configuration--No. 2	V-3
Alternative Tankage Configuration--No. 3	V-4
Alternative Tankage Configuration--No. 4	V-5
6.7×10^6-lb Plug-Nozzle Thrust Chamber Assembly	V-6
Thrust Chamber Assembly--Film-Cooled, Pressure-Fed Plug-Nozzle Engine	V-7
Plug Nozzle--6.7×10^6-lb Thrust, 32 Discrete Throats	V-8
K_w Versus Valve Weight	V-9
Thrust-Chamber Valves for 6,700K-lb-Thrust Pressure-Fed Engine	V-10
Actuation Torque Requirements for Thrust Chamber Valve	V-11
Thrust Chamber Valves K_w Versus Percent of Valve Opening	V-12

FIGURE LIST (cont.)

	FIGURE NO.
Acceleration Head During Booster Flight	V-13
Static Tank Pressure Requirements During Booster Flight	V-14
Fuel Tank Pressurization--Heated Hydrogen Pressurized by Heated Helium	V-15
Oxidizer Tank Pressurization--Heated Helium	V-16
Integrated Pressurization System	V-17
Burst-Diaphragm Valve	V-18
Helium-Pressure Control Valve	V-19
Alternative Pressurization System	V-20
Fuel-Tank Pressurization--Heated Hydrogen Pressurized by Heated Vaporized Hydrogen	V-21
Booster System With Vaporization of Main Propellants for Pressurization	V-22
Fuel Tank Pressurization--Exhaust Gases from Liquid Bipropellant Gas Generator	V-23
Oxidizer Tank Pressurization--Heated Oxygen	V-24
Oxidizer Tank Pressurization--Exhaust Gases from Liquid Bipropellant Gas Generator	V-25
VāPak Pressurization System--6×10^6-lb Thrust	V-26
Properties of Saturated O_2	V-27
Oxidizer Tank Pressurization--Heated Nitrogen	V-28
Thrust Versus Duration for Engine Operation on Tank Pressurization Gases, Pressure-Fed System	V-29
Control Moment for 90° Sector by Chamber Pressure Modulation	V-30
Manually Connected Fill or Vent Valve Coupling	V-31
Combination Topping Vent and Pressure Relief Valve	V-32
Primary Hydrogen-Tank Boiloff as Function of Tank Insulation Thickness, Frost Neglected	V-33
Helium Storage System	V-34

I. OBJECTIVE

The objective of this investigation was to provide design data for a large, pressure-fed, liquid-propellant, rocket booster stage. This system will be used as the basis of comparison for other competitive systems.

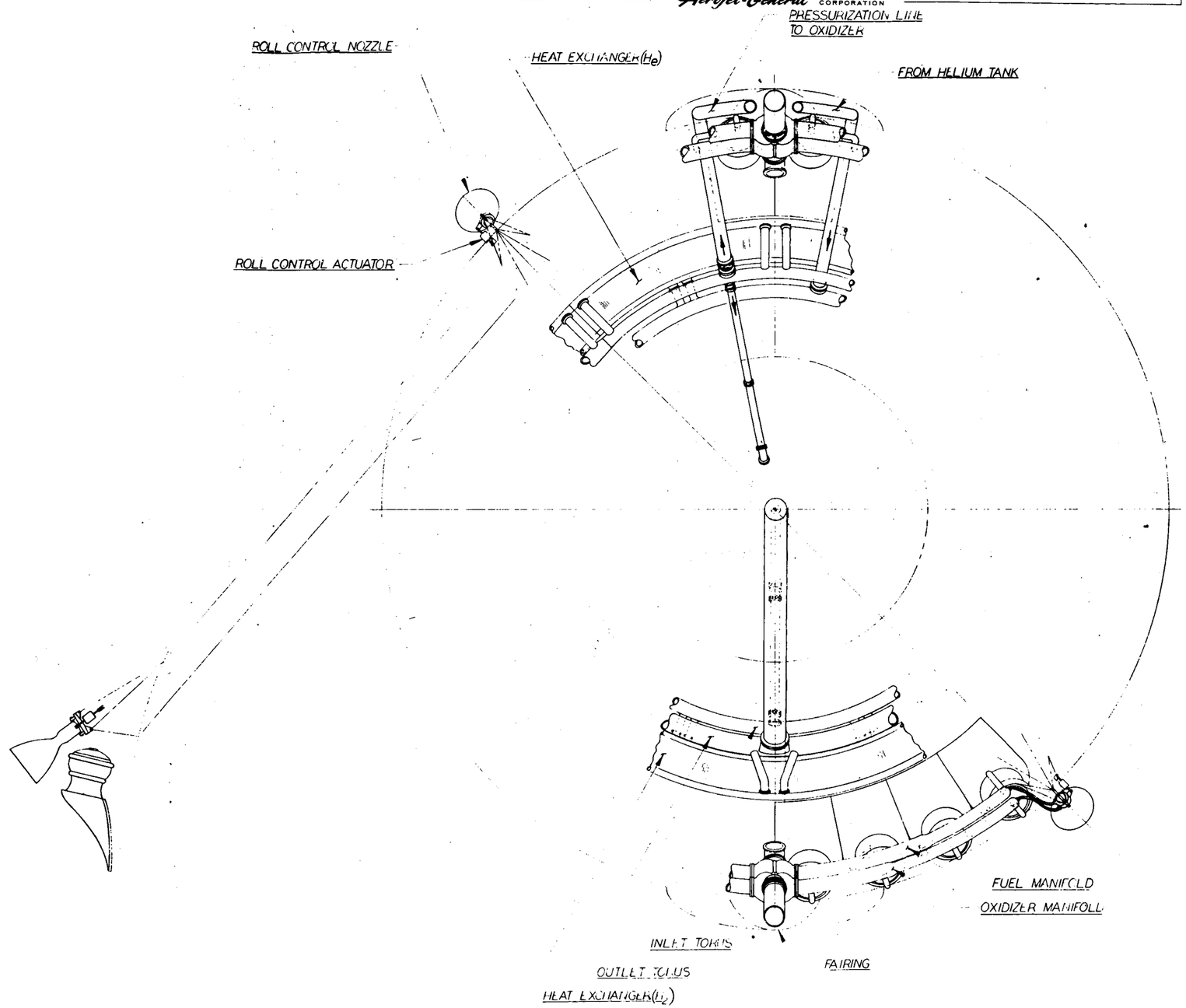
II. SUMMARY

The pressure-fed booster configuration evolved from this design study is pictured in Figure II-1, and shown schematically in Figure II-2. It was selected as the standard pressure-fed configuration after integration of the results of vehicle and subsystem design and analysis studies. These investigations included trajectory analysis; booster-performance analysis; engine design; propellant-tankage and structure design; propellant-tank pressurization-system design; thrust-vector and roll-control system design and analysis; and propellant-loading system design. Booster studies previously conducted by Aerojet-General and by other companies were also reviewed and evaluated. The Cosmos booster system study, conducted by the Aerojet-General Systems Division, was particularly useful because of the similarity in booster-design requirements.

The booster was designed to boost the same stage payload at an ideal velocity increment of 10,000 ft/sec as a 6,000,000 lb-thrust pump-fed booster. Although this velocity increment is not optimum for a pump-fed system, it is near optimum for this configuration.

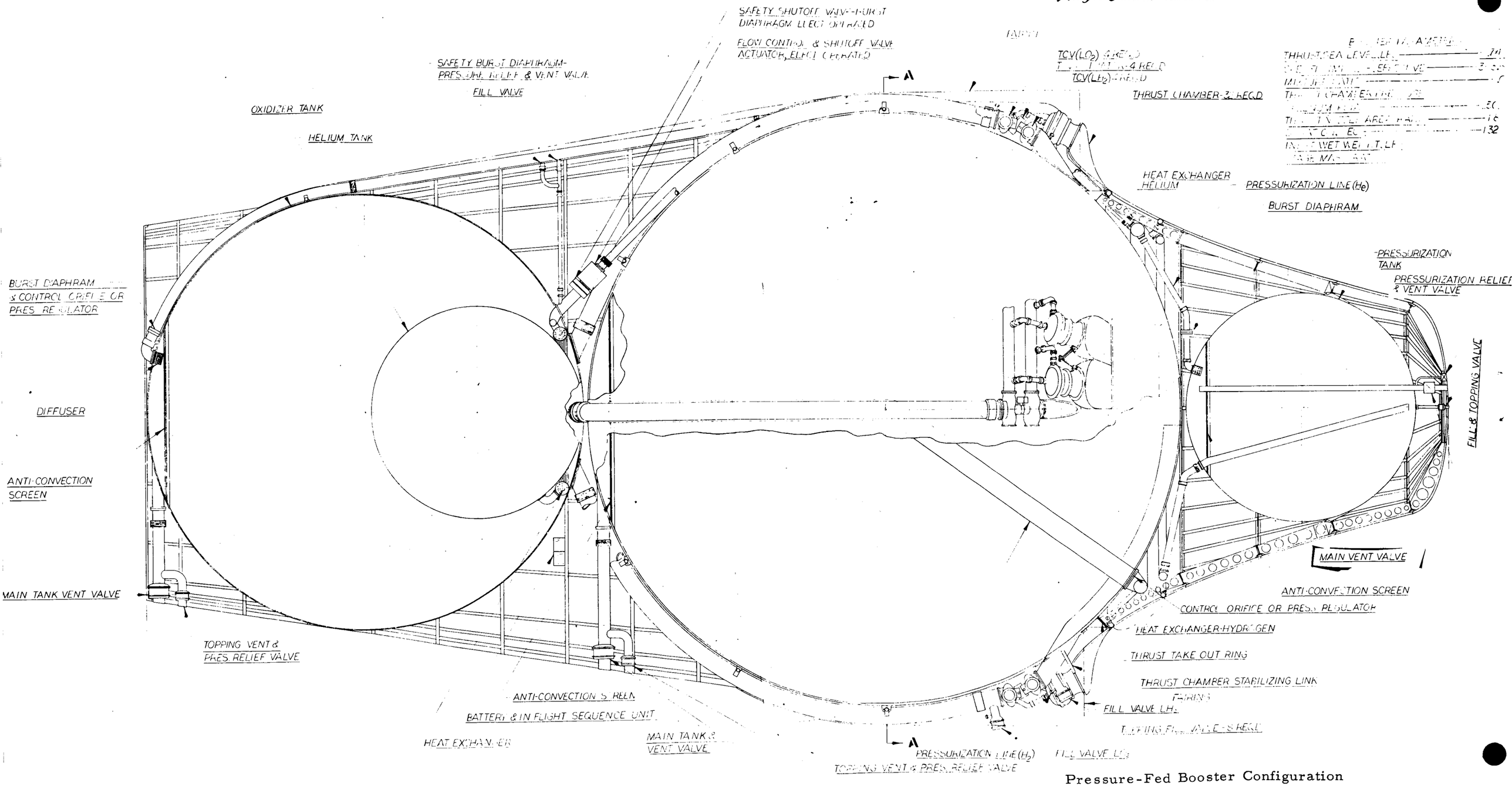
This booster has a plug nozzle engine which develops 6,700,000 lb of thrust at sea-level, using liquid oxygen and liquid hydrogen as propellants. Thirty-two separate film-cooled, pen-shaped thrust chambers are located around the engine expansion cone circumference and comprise the thrust-chamber subassembly. These chambers are operated at chamber pressure of 250 psia and mixture ratio of 6:1. Maximum expansion ratio is 18:1 and effective I_s is 364 sec.

Propellants are contained in single spherical tanks; the oxidizer tank located above the fuel tank. The oxidizer tank is pressurized by helium gas, which is heated by a heat exchanger located on the engine plug. Fuel-tank pressurization is provided by hydrogen stored in liquid form and heated in another heat exchanger. Heated helium gas is used to provide the pressure head for the hydrogen system.



Pressure-Fed Booster Configuration

Figure II-1
Sheet 1 of 2



Pressure-Fed Booster Configuration

Figure II-1
Sheet 2 of 2

23 ENGINE PLUG
WEIGHT 16,688 LB

24 PRESSURE RELIEF & VENT VALVE HIGH PRESSURE FILL
OPERATING PRESS
WEIGHT

25 HYDROGEN PRESSURIZATION TANK
DIA 23.91 FT
MATERIAL 2014 T6 OR AL ALLOY
WT EMPTY 11,000 LB
INITIAL H2 WT 12,736 LB
RESIDUAL H2 WT 1,896 LB

26 HEAT EXCHANGER (H_2)
MATERIAL CRES 347 TUBING
WT EMPTY 1018 LB
FLOW RATE 118.8 LB/SEC
ΔP 80 PSI

27 HEAT EXCHANGER (H_2)
MATERIAL CRES 347 TUBING
WEIGHT 3214 LB
FLOW RATE
ΔP 50 PSI

28 TOPPING FILL VALVE AUTO DISCONNECT

29 FILL CONNECTOR HIGH PRESSURE
DIA 6.00 IN
WEIGHT 32 LB
CAPACITY 600 GAL/MIN

30 BURST DIAPHRAGM

31 BURST DIAPHRAGM

32 ROLL CONTROL ACTUATOR ELECT OPERATED

33 ROLL CONTROL ENGINE

34 TOPPING FILL VALVE AUTO DISCONNECT
DIA 1.00 IN
WEIGHT 1.25 LB
CAPACITY 30 GAL/MIN

35 TOPPING FILL VALVE AUTO DISCONNECT
DIA 1.00 IN
WEIGHT 1.25 LB
CAPACITY 300 GAL/MIN

36 TOPPING VENT & PRESSURE RELIEF VALVE

37 HELIUM PRESSURE RELIEF & VENT VALVE

22 THRUST CHAMBER
THRUST 312,932 LB
P_c 324 PSIA
WEIGHT 811 LB
MR 6.11
FLOW RATE 6,593 LB/SEC
I_{sp} 307 SEC

21 ROLL CONTROL ENGINE
THRUST 32,300 LB
P_c 340 PSIA
WEIGHT 190 LB
MR 2.0
FLOW RATE 668 LB/SEC
I_{sp} 295 SEC

20 ROLL CONTROL ACTUATOR ELECT. OPERATED
HORSEPOWER
WEIGHT

19 PRESSURE RELIEF & VENT VALVE (ROLL CONTROL)
BURST PRESS

18 THRUST CHAMBER VALVE
QUANTITY 4
FLOW RATE 668 LB/SEC
ΔP 9 PSI
WEIGHT 439 LB

17 THRUST CHAMBER VALVE
QUANTITY 4
FLOW RATE 668 LB/SEC
ΔP 9 PSI
WEIGHT 439 LB

16 TCV ACTUATOR ELECT. OPERATED
QUANTITY 12
WEIGHT 1210

15 FILL CONNECTION
WEIGHT 32 LBS
CAPACITY 600 GAL/MIN

14 ELECT. POWER SUPPLY BATTERY
SILVER CELL PRIMARY (NON RECHARGEABLE)
VOLTAGE 28 V
POWER SUPPLY 2.4 KW
WEIGHT 240 LB

13 FLOW CONTROL & SHUTOFF VALVE ACTUATOR, ELECT. OPERATED
HORSEPOWER
WEIGHT

12 SHUTOFF VALVE - BURST DIAPHRAGM ELECT OPERATED
WEIGHT

11 FILL CONNECTION
DIA 1.00 IN
WEIGHT 1.25 LB
CAPACITY 300 GAL/MIN

10 FILL CONNECTION
DIA 10.00 IN
WEIGHT 29 LB
CAPACITY 2,000 GAL/MIN

9 HELIUM TANK BURST DIAPHRAGM
BURST PRESS 4320 PSIA

8 H_2 PRESSURIZATION TANK
DIA 21.08 FT
PRESS 3370 PSIA
MATERIAL 2014-T6 ALUMINUM
WEIGHT 19,300 LB
INITIAL H_2 WT 2,510 LB
RESIDUAL H_2 WT 5,895 LB

7 PRESSURE RELIEF & VENT VALVE
OPERATING PRESS 368 PSIA
MAX FLOW RATE 391 LB/SEC
WEIGHT 30 LB

6 BURST DIAPHRAGM & CONTROL ORIFICE OR PRESSURE REGULATOR

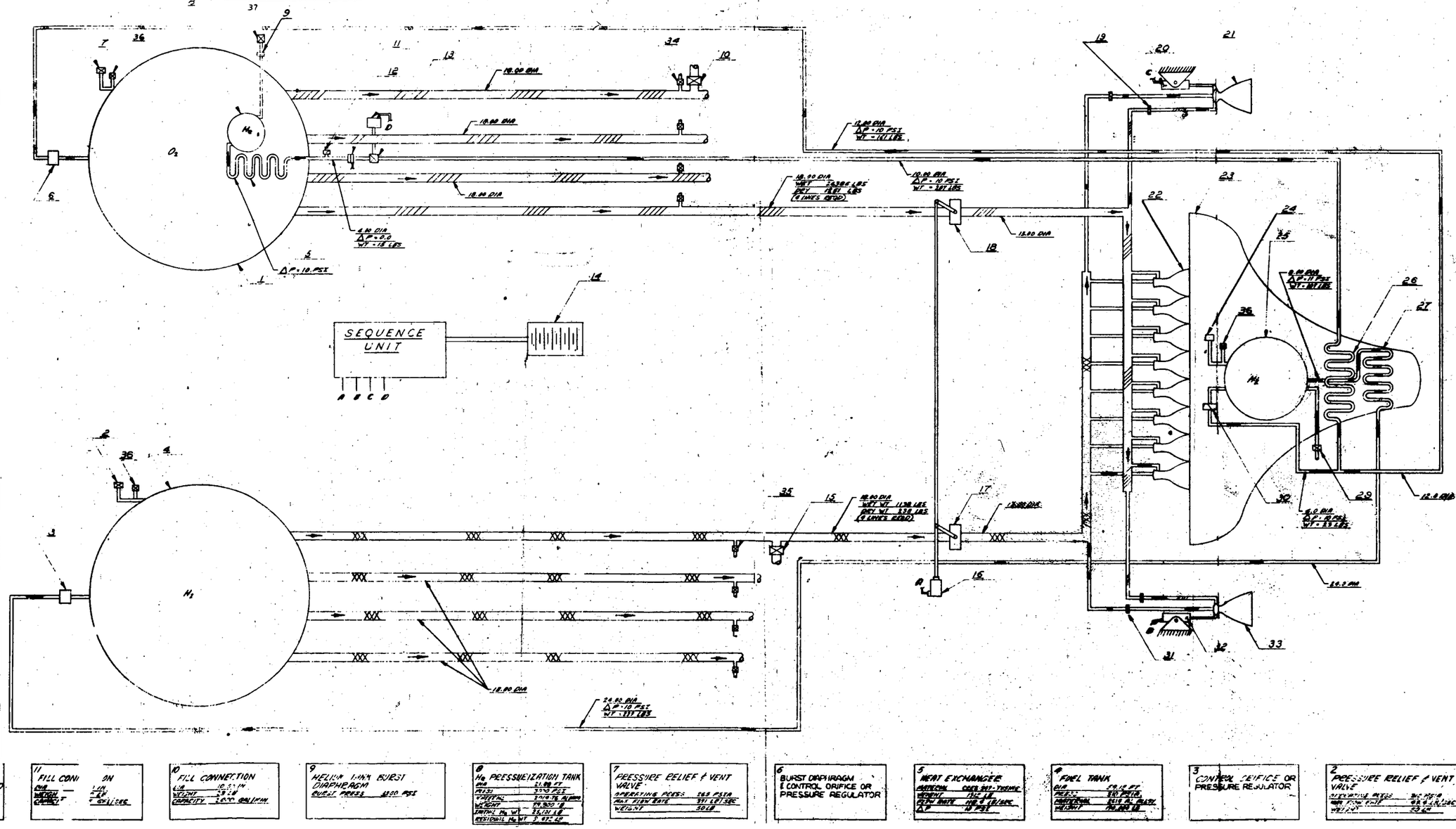
5 HEAT EXCHANGER
MATERIAL CRES 347 - TUBING
WEIGHT 132 LB
FLOW RATE 118.8 LB/SEC
ΔP 10 PSI

4 FUEL TANK
DIA 23.8 FT
PRESS 350 PSIA
MATERIAL 6AL4 AL ALLOY
WEIGHT 740 LB

3 CONTROL ORIFICE OR PRESSURE REGULATOR

2 PRESSURE RELIEF & VENT VALVE
TANK OPERATING PRESS 364 PSIA
MAX FLOW RATE 68.4 LB/SEC
WEIGHT 20 LB

1 OXIDIZER TANK
DIA 50.08 FT
PRESSURE 365 PSIA
MATERIAL 6AL4 AL ALLOY
WEIGHT 77,300



Schematic, Pressure-Fed Booster Configuration

Figure II-2

III. PHILOSOPHY AND SCOPE

System studies of large boosters, including pump-fed and pressure-fed engines, have been conducted by Aerojet-General and by other companies.

These investigations were directed toward evolving an optimum system configuration and were based on total system cost or cost per pound of payload. However, the relative operational complexity of various vehicle configurations was not considered in detail. Therefore, this investigation was undertaken to establish the functional components required for a large pressure-fed booster. Component operating characteristics as well as fabrication and development problems would also be determined. The resulting configuration could then be compared with a pump-fed booster of similar size.

Previously conducted investigations have provided near-optimum configurations for nonfunctional components; therefore, the selection of tank material, structural design, and tank shape was based on the results of these concluded study efforts.

Wherever possible, considering the change in scale, fully proved concepts were used so that the configuration could be immediately manufactured.

Liquid oxygen and liquid hydrogen were selected as the propellants.

IV. BOOSTER SPECIFICATIONS

A. PARAMETER SELECTION

The parameters selected for the pressure-fed booster investigation include the following:

Ideal velocity increment (ft/sec)--10,000
Thrust (lb)--6,700,000
Propellant mixture ratio--6.0:1
Thrust chamber pressure (psia)--250
Nozzle expansion ratio--18:1

Each of these parameters is discussed individually in this section.

1. Booster Ideal Velocity Increment

The ideal velocity increment is required to calculate propellant tank volume. Final selection of booster ideal velocity increment depends upon vehicle mission, the number of stages, the performance of each individual stage, and vehicle and support costs. Only approximate values for most of these variables are available. Therefore, it was not possible, within the scope of this study, to establish an optimum value for the booster ideal velocity increment. However, previous studies have shown that the ideal velocity increment should be between 8,000 and 12,000 ft/sec for a pressure-fed booster. As an example, Figure IV-1 shows the 300 nm orbit vehicle payload as a function of booster ideal velocity increment for a two-stage vehicle using the LOX/LH_2 pressure-fed booster and LOX/LH_2 pump-fed upper stage. For maximum payload, a booster ideal velocity increment of 10,000 would be selected. The results of another study in which the relative costs of pump-fed and pressure-fed systems were

IV, A, Parameter Selection (cont.)

analyzed are shown in Figures IV-2 through -4. Note that above 10,000 ft/sec of booster ideal velocity increment the relative cost of pressure-fed boosters increases rapidly; therefore, a booster ideal velocity increment of 10,000 ft/sec was selected for this investigation.

2. Thrust Level

A sea-level thrust of 6,700,000 lb was selected. Using this thrust level and a vehicle thrust-to-takeoff weight ratio of 1.25:1, the pressure-fed booster has a stage payload capability of approximately 2-million-lb for an ideal velocity increment of 10,000 ft/sec. Thus, the pressure-fed booster has approximately the same stage payload capability as a booster using the standard 6-million-lb thrust pump-fed LOX/LH_2 engine with the same ideal velocity increment and initial thrust-to-weight ratio. The vehicle payload would be greater using the pump-fed booster because the optimum pump-fed booster ideal velocity increment would be in excess of 15,000 ft/sec.

3. Mixture Ratio Selection

There are three major considerations that influence the final selection of mixture ratio in a pressure-fed booster using LOX/LH_2 propellants. These are specific impulse, tank weight, and combustion temperature. The mixture ratio giving maximum specific impulse is approximately 4.5:1; however, at higher mixture ratios, a significant savings in propellant volume can be effected with very little specific impulse loss because propellant bulk density is higher. On a cost basis, considering only these effects, the best mixture ratio may be as high as 8 or 10:1. As mixture ratio is increased, however, combustion temperature and consequently combustion chamber heat-transfer rates also

IV, A, Parameter Selection (cont.)

increase. In a low-pressure nozzle, this can result in extremely severe cooling problems. Thus, a complete analysis to select optimum mixture ratio would not be feasible until the heat-transfer requirements are known. A mixture ratio of 6:1 was selected for this design study because it results in near-maximum payload.

4. Thrust-Chamber Pressure

The thrust-chamber pressure was selected on the basis of the optimum cost study discussed previously. In this study, pressure-fed boosters using thrust-chamber pressures from 100 to 550 psia were evaluated. Pertinent study results are presented in Figure IV-4. Minimum cost per pound of booster payload is obtained over most of the velocity range investigated at a thrust-chamber pressure of approximately 250 psi. Hence, this pressure was selected for the pressure-fed booster studied.

5. Nozzle Area Ratio

The sea-level specific impulse of a plug-nozzle engine is almost completely independent of area ratio. Vacuum specific impulse improves with increasing area ratio. Consequently, the highest average specific impulse is obtained when the maximum available area ratio is used. At a fixed chamber pressure, the area ratio obtainable is limited by the vehicle diameter. An area ratio of 18:1 was obtained as a result of the vehicle design study.

(End IV-A.)

IV, Booster Specifications (cont.)

B. VEHICLE ANALYSIS

Physical and performance parameters of a vehicle using the pressure-fed booster are presented in Table IV-1. The selection of an upper stage was necessary to establish thrust-vector-control requirements and to determine vehicle orbital-payload capability. Booster propellant requirements which dictate tank sizes are also delineated. The vehicle acceleration through booster flight is shown in Figure IV-5. Acceleration forces were considered when determining pressurization requirements.

The exchange ratio between specific impulse and inert weights is 4,900 lb/sec and was determined as shown in paragraph 3 of this section.

1. Trajectory

Trajectory calculations were performed to determine the 300 nm circular orbit payload capability of a 6,700,000 lb sea-level thrust pressure-fed O_2/H_2 booster with an O_2/H_2 pump-fed second stage. Results and assumptions used in the calculations are summarized in Table IV B-1 and the vehicle trajectory is described below.

a. The vehicle rises vertically with zero angle of attack until a velocity of 280 ft/sec is attained. At this velocity, a step-change in the angle of attack (commonly termed kick angle) is assumed. The value of the kick angle is determined so that the magnitude and direction of the velocity vector necessary for circular orbit are achieved simultaneously. For the vehicle described in Table IV-1, this kick angle is 5.37 degrees. The altitude at this point on the trajectory is approximately 3,400 ft.

IV, B, Vehicle Analysis (cont.)

b. After the kick angle, the vehicle attitude remains constant until the angle of attack returns to zero. A zero angle of attack is then maintained throughout the remaining powered phase to low altitude orbit. This is referred to as a gravity turn trajectory.

c. The vehicle is powered continuously except for staging, until low altitude orbit (commonly termed a "parking orbit") is achieved. The altitude of this parking orbit depends upon the vehicle configuration; stage thrust levels, specific impulses, etc. For this vehicle, the altitude of the parking orbit is approximately 60 nm.

d. The vehicle is transferred from the low altitude orbit to the desired orbit (300 nm) on a Hohmann transfer ellipse.

Trajectory calculations performed by numerically integrating the equations of motion in the IBM 7090 computer have shown that the maximum vehicle payload capability is obtained by following the previously described trajectory.

Other assumptions used in performing the trajectory analysis are as follows:

(1) Launch is assumed to be eastward from Cape Canaveral.

(2) A rotating-earth correction is applied at first-stage burnout.

(3) Staging is instantaneous.

IV, B, Vehicle Analysis (cont.)

2. Thrust-Vector Control Requirements

Thrust-vector control requirements were determined by considering aerodynamic stability requirements, guidance requirements, and correction for thrust-vector misalignment. The following requirements were established. These values are the maximum required for the assumed conditions. Modification to the assumed "kick angle" duration as discussed under "Guidance" would allow a smaller thrust-vector control angle.

Source	Necessary Control Moment	Equivalent Thrust Vector Angle
Aerodynamic Stability	None	None
Guidance	11.2×10^6 Ft-Lb	1.0°
Alignment Tolerances	5.6×10^6 Ft-Lb	0.5°
RMS TOTAL	12.5×10^6 Ft-Lb	1.12°

a. Aerodynamic Stability

The configuration selected for this study (Figure IV-6) has desirable aerodynamic characteristics. The uncontrolled vehicle is statically stable (i.e., the center of gravity is forward of the center of pressure) by a small margin. Consequently, the vehicle responds to disturbing airloads in a manner which reduces the magnitude of these loads. The natural frequency of the short period pitching motion of the uncontrolled vehicle is approximately 0.1 cps.

IV, B, Vehicle Analysis (cont.)

Changes in the margin of static stability can be made by modification of the nose cone profile. The value of C_{M_α}, the pitching moment coefficient slope, for example, changes from -0.10 to -0.20 when a conical nose half angle changes from 26.5° to 45°. The aerodynamic moment, being proportional to C_{M_α}, changes by a factor of two when this change in the nose cone profile is made. Because this configuration is near the neutral stability point, its stability requirements are sensitive to small changes in profile. It can be a desirable range to work in, however, because response rate requirements are low in this region. Whether stable or unstable (either is possible with this configuration) the pitching motion rate will be low, allowing low response rate requirements for the control system.

b. Guidance

Guidance requirements for the thrust-vector control system are dependent upon the desired trajectory. The trajectory assumed in this study applies a "kick angle" of 5.37°. For trajectory study, it is assumed that a step change in angle of attack of 5.37° is applied at a prescribed time. Of course, in an actual vehicle there must be some time associated with accomplishment of the "kick". The details of vehicle response during this maneuver are investigated in the following paragraphs. It will be assumed that vehicle attitude changes as described in Figure IV-7 where:

$$\theta = \frac{A}{2}(1 - \cos wt)$$

$$\dot{\theta} = \frac{A}{2} w \sin wt \qquad 0 \leq t \leq t_1$$

$$\ddot{\theta} = \frac{A}{2} w^2 \cos wt$$

IV, B, Vehicle Analysis (cont.)

In the above equations:

$$w = \text{angular velocity} = 2\pi f$$

$$\text{where } f = \frac{1}{2t_1} \text{ cps.}$$

$$\therefore w = \frac{\pi}{t_1}$$

Immediately before the "kick" maneuver angle of attack α and aerodynamic moment are zero. Therefore, initial $\ddot{\theta}$, which is also maximum $\ddot{\theta}$, must be caused entirely by control moment. At this instant the following equations apply.

$$I\ddot{\theta}_{MAX} = F\ell\beta_{MAX}$$

$$\therefore \quad \beta_{MAX} = \frac{I\ddot{\theta}_{MAX}}{F\ell}$$

$$\ddot{\theta}_{MAX} = \frac{A}{2} w^2 = \frac{A}{2} \frac{\pi^2}{t_1^2}$$

$$\therefore \quad \beta_{MAX} = \frac{I\pi^2}{2F\ell} \left(\frac{A}{t_1^2} \right)$$

IV, B, Vehicle Analysis (cont.)

where

I = Vehicle moment of inertia, slug - ft^2

$\ddot{\theta}_{MAX}$ = Max. vehicle angular acceleration, rad/sec^2

F = Thrust

ℓ = Length from equivalent gimbal pt. to vehicle c.g.

β_{MAX} = Max. thrust vector angle

A = "Kick" angle

t_1 = Duration of "kick" maneuver

The above equation for β_{MAX} has been plotted in Figure IV-8 at t = 25 sec after launch.

at t = 25 sec

h = 3,400 ft

v = 280 ft/sec

I = 515 slug - ft^2

F = 6.6×10^6 lb

ℓ = 106 ft

A thrust-vector requirement selected from Figure IV-8 indicates that β_{MAX} = 1° allows the accomplishment of a 4° kick in 4 sec or a 6° kick in 5 sec. Thus, a 1° thrust-vector angle appears adequate for the kick maneuver. It is also indicated that with a longer kick duration this maneuver may be accomplished with a smaller thrust-vector angle.

IV, B, Vehicle Analysis (cont.)

From Figure IV-7, it is seen that $\ddot{\theta}$ rises from zero to a maximum value instantaneously. Practically, the time to accomplish this rise is limited by thrust-vector control system response capability. It is reasonable to expect that this control moment may be applied in one sec. Thus, the total time to accomplish the kick would be increased by some what less than one sec.

The control moment necessary to maintain the angle of attack after the kick maneuver may be calculated as follows:

$$m = C_{M\alpha} \alpha q \, SL$$

m = aerodynamic moment
$C_{M\alpha}$ = pitching moment coefficient slope = 0.20
α = angle of attack = 5.37° = 0.094 rad.
q = dynamic pressure = 86.2 lb/ft^2
S = frontal area - 2640 ft^2
L = reference length - 225 ft.

$\therefore$ $m = (0.20)(0.094)(86.2)(2,640)(225) = 960,000$ ft-lb

The 0.96×10^6 ft-lb moment necessary to maintain a 5.37° angle of attack is insignificant when compared to the 11.2×10^6 ft-lb moment necessary to perform the kick maneuver.

Following the kick maneuver, some slight vehicle attitude change is necessary to maintain the vehicle center line tangent to the flight path. The maximum vehicle angular acceleration derived from this requirement is 0.018 degrees/sec^2. This angular acceleration requires a control moment of 0.16×10^6 ft-lb. This is again a negligible requirement.

IV, B, Vehicle Analysis (cont.)

c. Alignment Tolerances

A value of 0.5° has been assumed for thrust-vector misalignment.

d. Roll Control Requirements

A roll control moment of 1.2×10^6 ft-lb was used in the referenced Cosmos Report and was also assumed for this vehicle.

3. Inert Weight - Specific Impulse Exchange Ratio

As an aid to vehicle component designs, the inert weight-specific impulse tradeoff was calculated. To calculate this, the following formula was derived from the expression for ideal velocity increment.

$$\frac{\Delta W_I}{\Delta I_{SP}} = (v)(W_{TO}) \left[\frac{e^{-\frac{V}{(gc)(ISPV)}}}{(gc)(ISPV)^2} \right]$$

ΔW_I = Change in inert weight, lbs.
ΔISP = Change in specific impulse, secs
v = Stage ideal velocity, ft/sec
W_{TO} = Stage lightoff weight, lbs
ISPV = Stage engine vacuum specific impulse, secs
gc = Force - Mass conversion factor, 32.174 ft/sec^2

For the booster described in Table IV-1,

v = 10,000 ft/sec
$W_{TO} = 5.36 \times 10^6$ lbs

IV, B, Vehicle Analysis (cont.)

$$ISPV = 390 \text{ sec}$$

$$\frac{\Delta M_I}{\Delta ISP} = 4{,}900 \frac{\text{lb}}{\text{sec}}$$

TABLE IV-1

300-NM ORBIT PERFORMANCE FOR A PRESSURE-FED BOOSTER WITH PUMP-FED SECOND STAGE (O_2/H_2 PROPELLANTS)

Parameter	1st Stage (1)	2nd Stage (2)
Thrust		
Sea Level	6.7×10^6	---
Vacuum	8.65×10^6	3.2×10^6
Engine Specific Impulse		
Sea Level	302	---
Vacuum	390	426
Lightoff Weight - Lbs	5,360,000	1,857,000
Useable Propellant Wt - Lbs	2,970,000	1,456,000
Burnout Weight - Lbs (3)	2,390,000	401,000
Jettison Weight - Lbs (4)	533,000	111,000
Payload - Lbs	---	290,000
Engine Propellant Weight Flow Rate (Lbs/Sec)	22,200	7,500
Duration - Secs	134	194
Ideal Velocity (Ft/Sec)	10,150	21,000
Useful Propellant Weight Ratio	.85	.93

(1) Plug Nozzle, Pc = 250 psia, A_e/A_t = 18/1, MR = 6.0 (92% of theoretical shift)

(2) DeLaval Nozzle, Pc = 1000 psia, A_e/A_t = 40/1, MR = 6.0 (94% of theoretical shift)

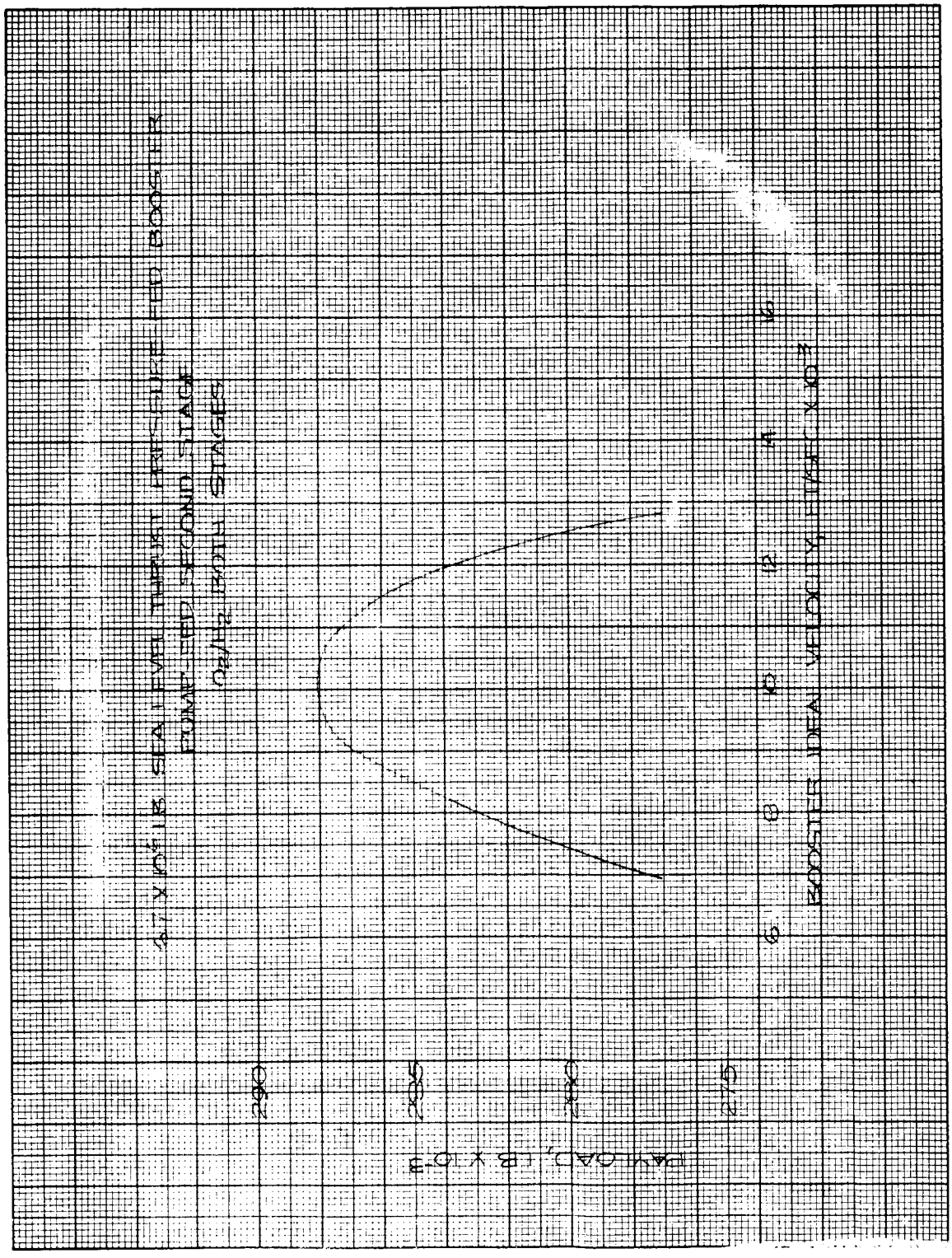
(3) Burnout weight includes stage inert weight, propellant outage of upper stages and payload

(4) Jettison weight includes stage inert weight and propellant outage

	First Stage	Second Stage
Engine sea level thrust, lb_f	6.7×10^6	-
Engine vacuum thrust, lb_f	8.85×10^6	Optimum
Thrust to lightoff mass ratio	1.25	Optimum
Engine vacuum specific impulse, secs	394	426
Thrust chamber pressure, psia	250	1,000
Area ratio	16.3:1	40:1
Propellant tank mass factor, lb/ft^3	2.5	.5
Propellant outage fraction	.01	.01
Propellant ullage fraction	.04	.04
Thrust dependent inert mass factor	.016	.02

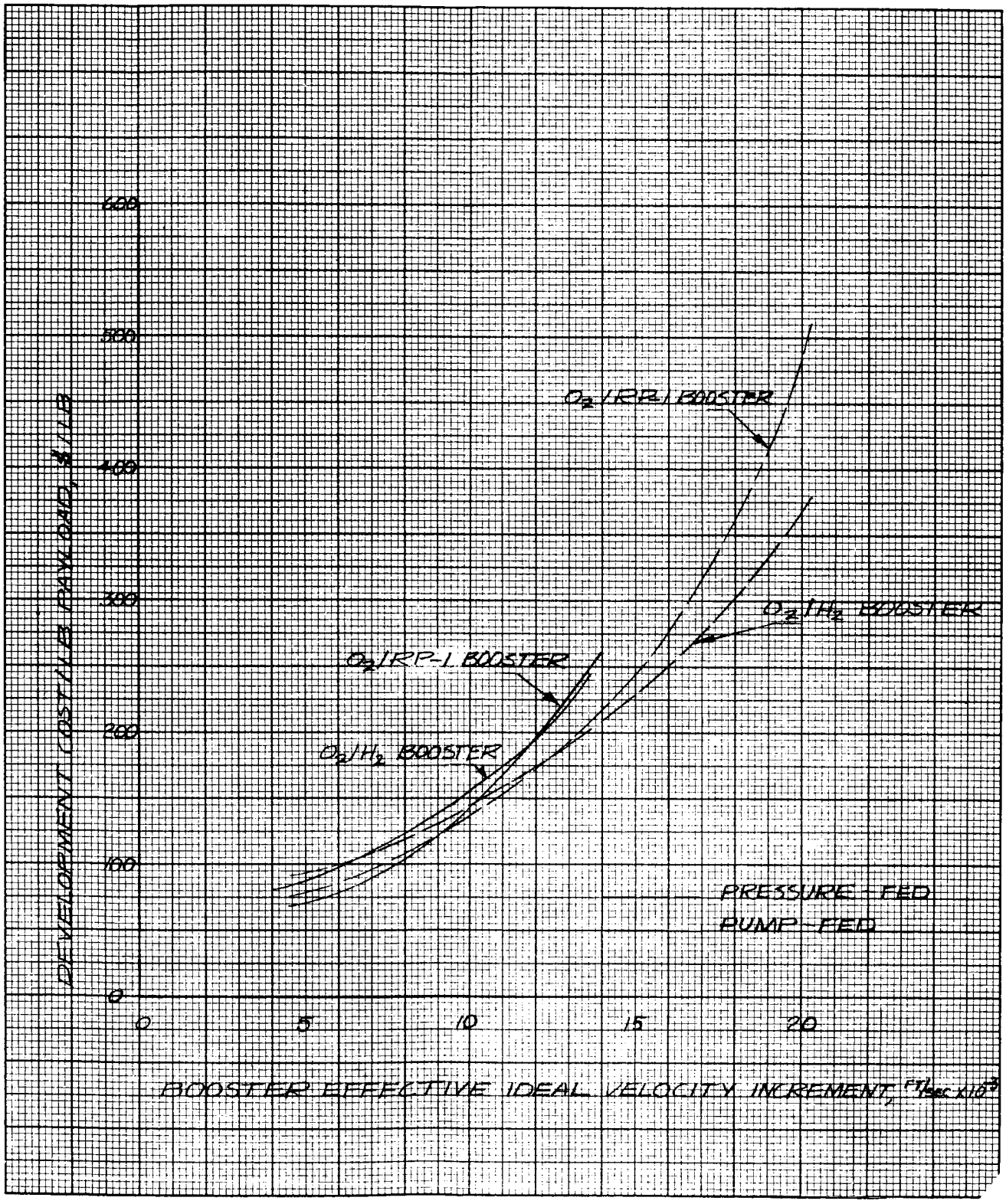
Figure IV-1
Page 1 of 2

Optimum Staging Investigation for a 300-NM-Orbit Mission



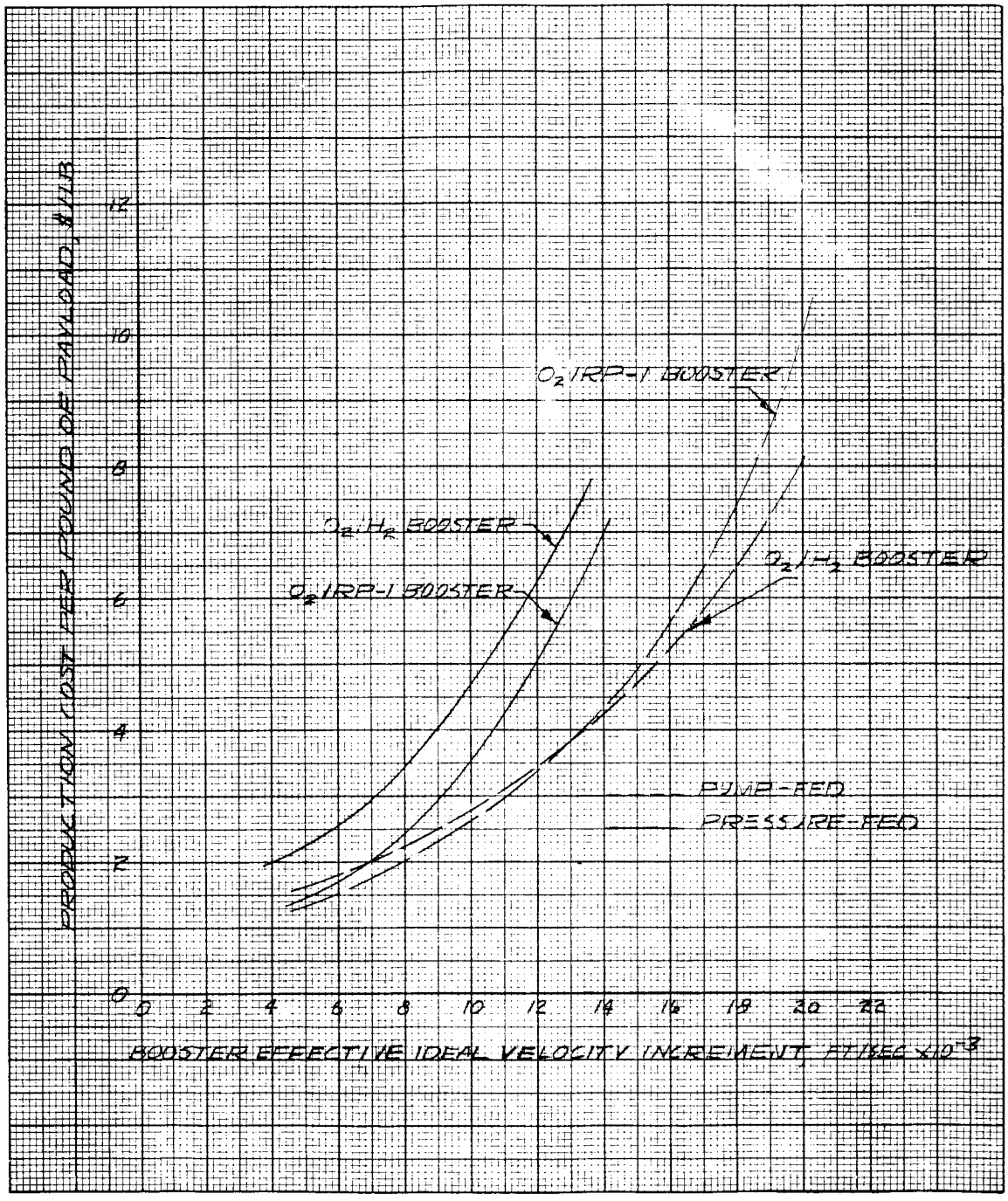
Optimum Staging Investigation for a 300-NM-Orbit Mission

Figure IV-1
Page 2 of 2



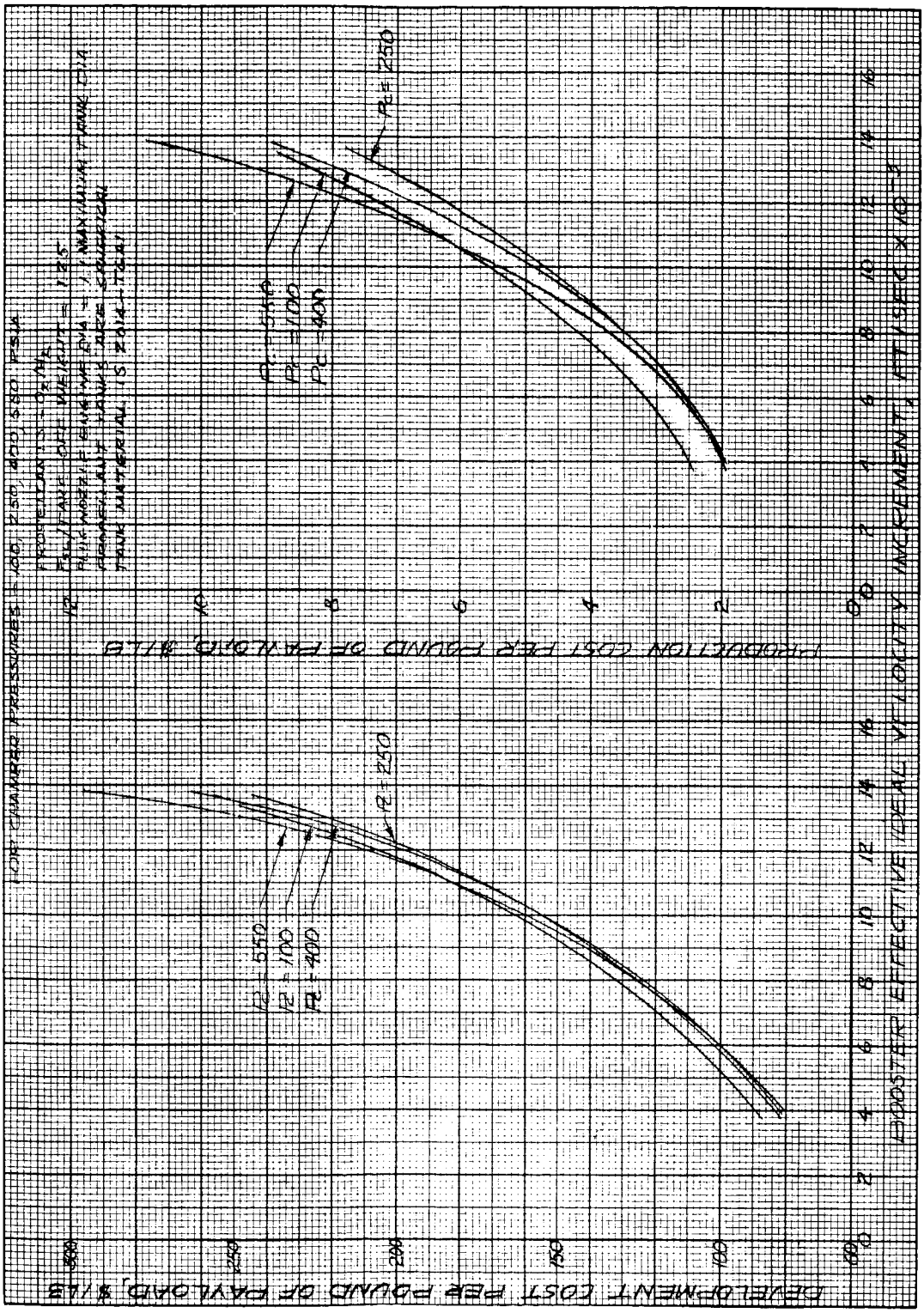
Booster Development Cost Per Pound of Payload Versus Booster Effective Ideal Velocity Increment

Figure IV-2



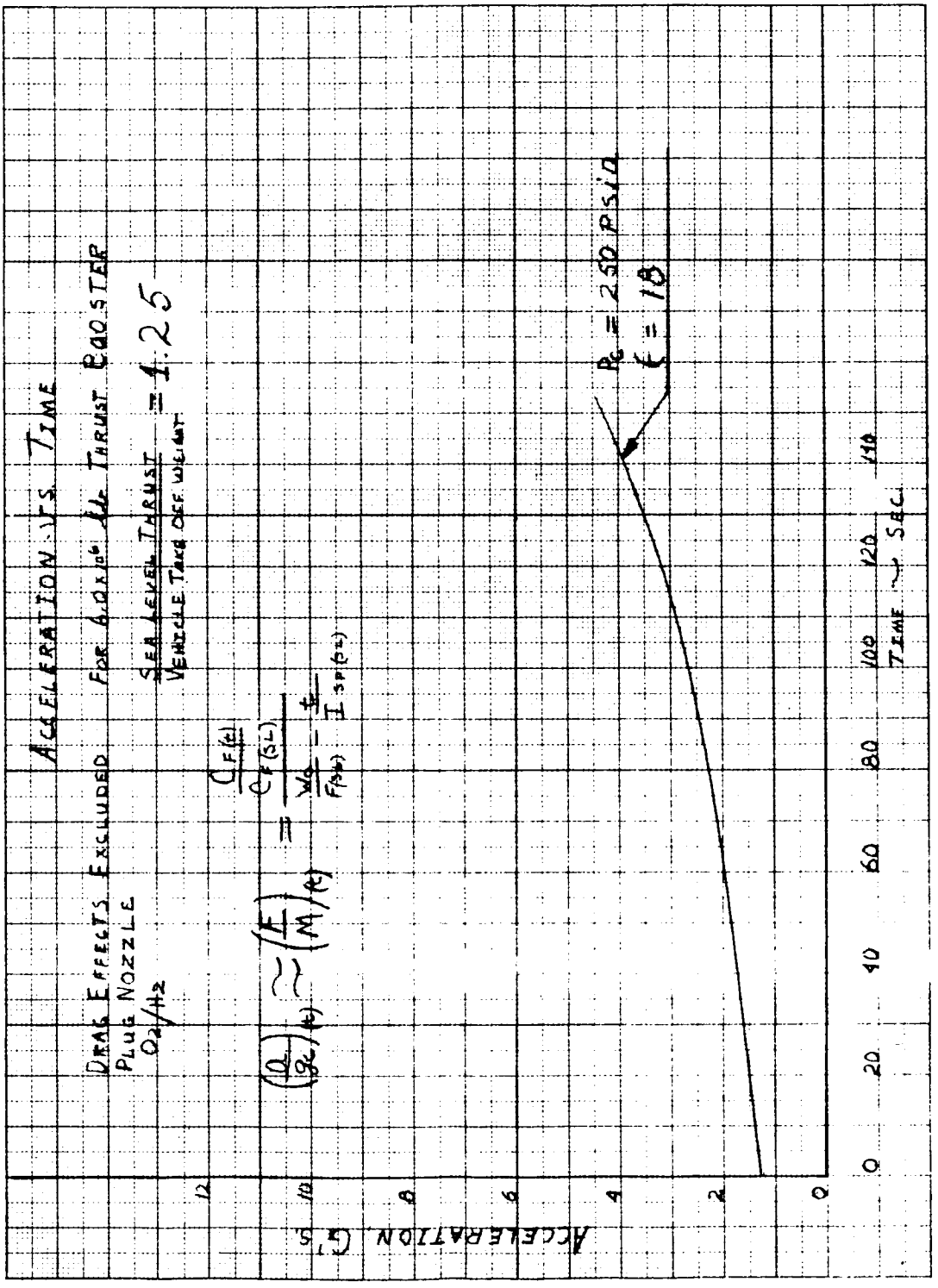
Booster Production Cost Per Pound of Payload Versus Booster Effective Ideal Velocity Increment

Figure IV-3

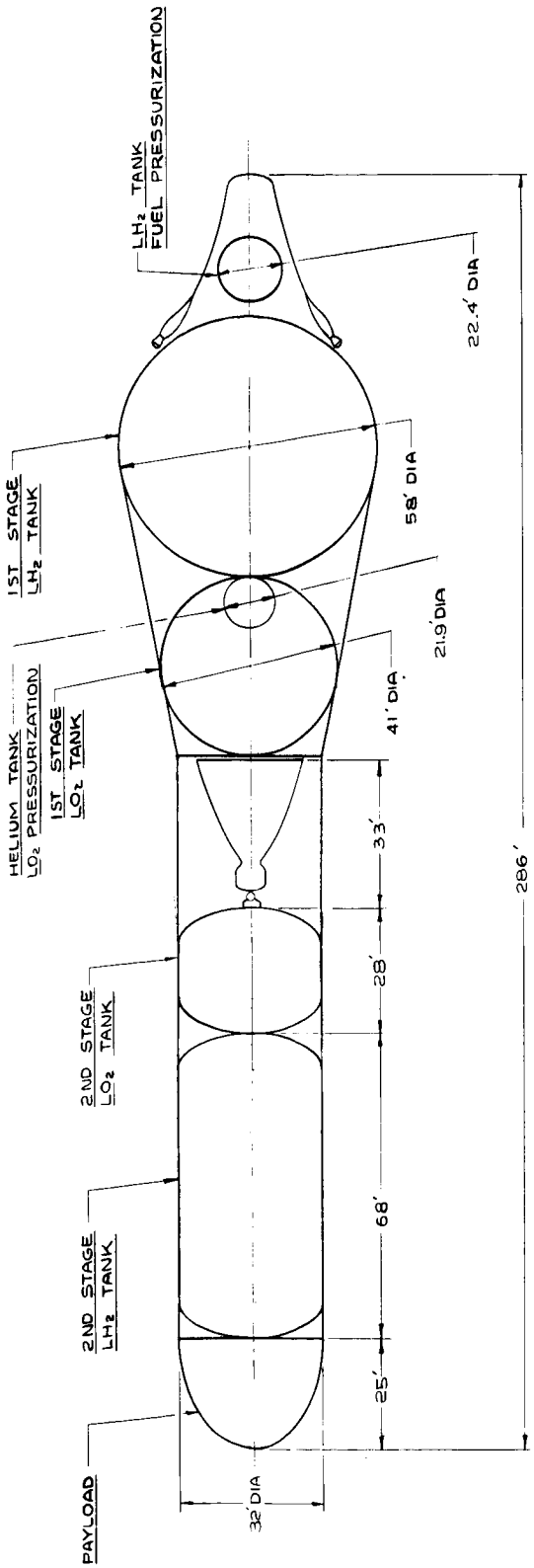


Pressure-Fed Booster Cost--Dollars Per Pound of Payload Versus Booster Effective Ideal Velocity Increment

Figure IV-4

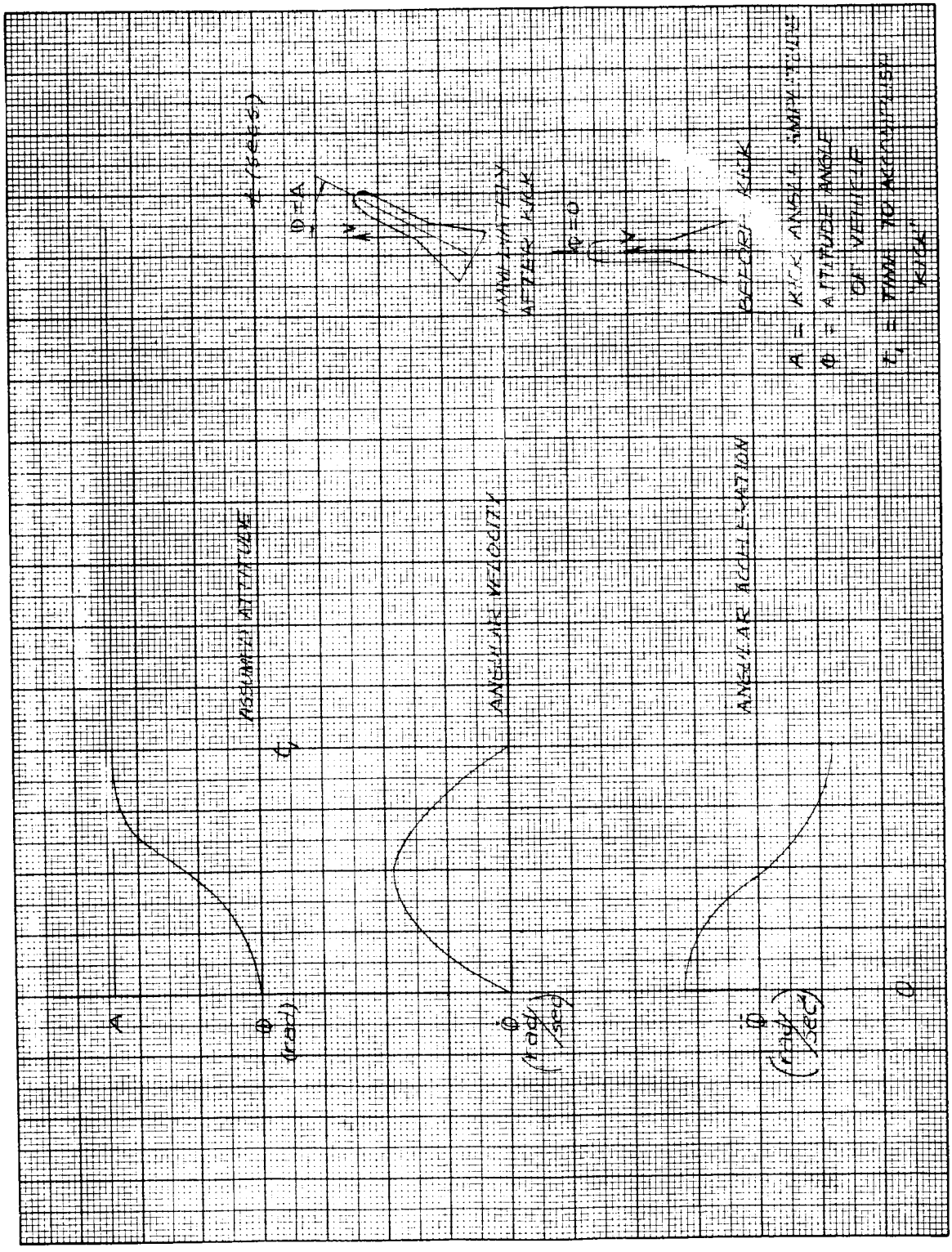


Acceleration Versus Time (6 x 10^6-lb-Thrust Booster)



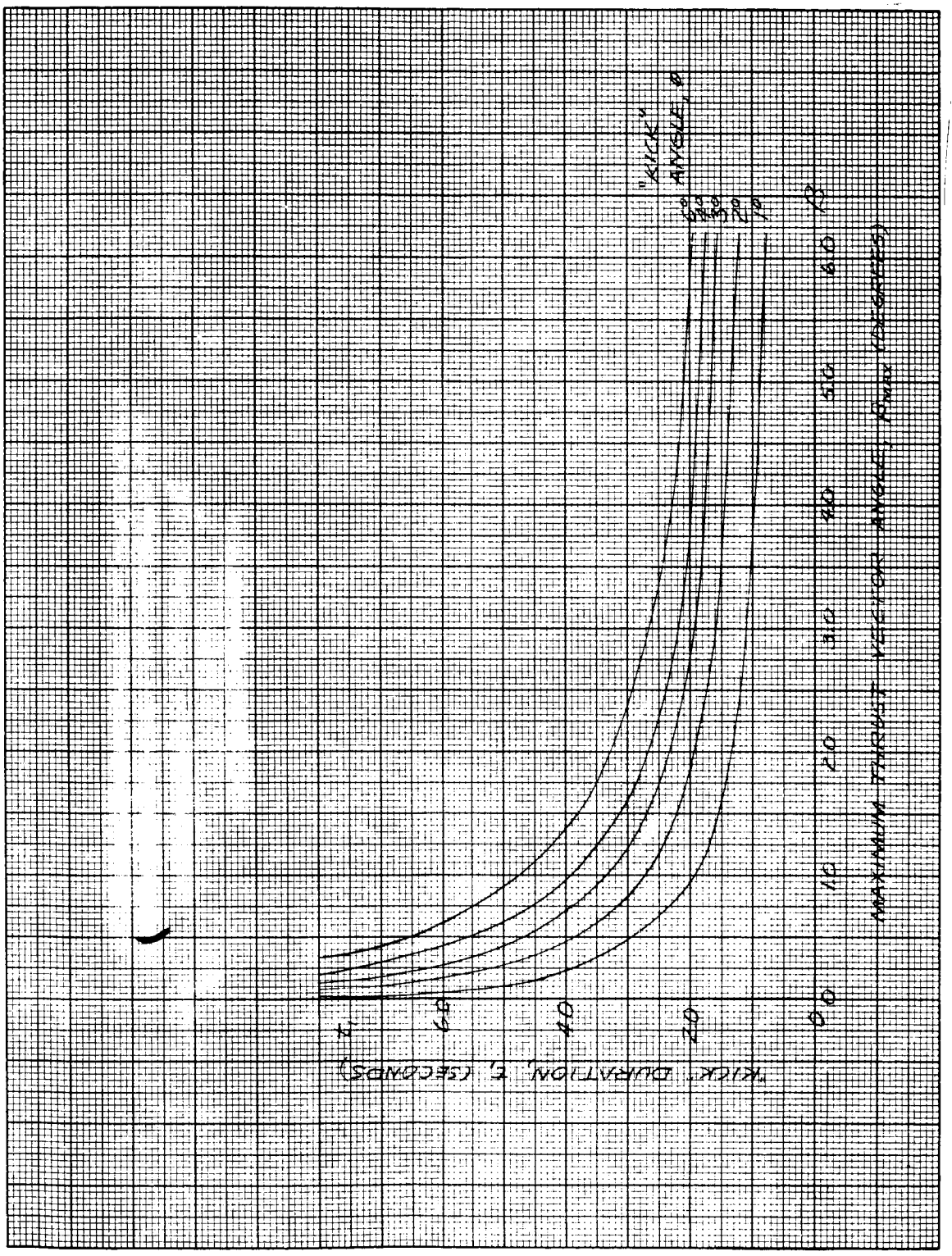
Conventional Pressure-Fed Vehicle, 6.7×10^6-lb Booster Thrust

Figure IV-6



Vehicle Response at Assumed Kick-Angle Application--Standard Pressure-Fed Vehicle, 6.7 x 10⁶-lb Thrust

Figure IV-7



Thrust-Vector Angle Necessary to Accomplish Trajectory Kick--Standard Pressure-Fed Vehicle, 6.7×10^6-lb Thrust

Figure IV-8

V. BOOSTER DESIGN STUDIES

A. TANKAGE STUDY

This study was for the purpose of determining if a reduction in weight and system complexity is possible when a common bulk-head and/or the plug space are used for propellant. Previous studies (i.e., Cosmos) have been conducted to develop an optimum tankage configuration and to select the best material. Results of the Cosmos study were used in selecting a material and a standard booster tankage configuration to provide a basis for comparison.

1. Shapes Considered

Different combinations of spheres, cylinders, and ellipsoids were considered and are shown in Figures V-1 through -5. The combinations are in two different categories; those which use the plug volume for propellant and those that do not.

Figure V-1 is a vehicle with two spherical tanks joined by stringer panel-rib intertank structures.

Tankage shown in Figure V-2 is a single sphere with the propellants separated by a common bulkhead. This configruation has the advantage of a reduction in overall vehicle length and a possible reduction in weight because intertank structure is eliminated.

Figures V-3 through -5 use the plug volume. Figure V-3 is similar to Figure V-1. Figures V-4 and V-5 are cylinders with spherical or ellipsoidal oxidizer tanks. The cylindrical fuel tanks were considered because of the ease of attaching the plug without a significant weight penalty.

V, A, Tankage Study (cont.)

2. Material Selection

In the Cosmos study conducted by this company, systems using both 2014 T6 aluminum and titanium alloys were evaluated.

The material selected for weight comparison in this study was 2014 T6 aluminum. There is considerable experience with welding 2014 plates up to 1/4-in. thickness. However, an investigation conducted by the Aerojet-General Corporation indicates that current technology can be extended to plates of 2 in. thickness, or greater, and that design concepts involving this fabrication process should not be ignored. Welding speeds of 15 min/ft are indicated at an approximate cost of $10,00 per foot, per inch of plate thickness.

Some advantages could be gained in the reduction of weight and fabrication complexity by using titanium alloys because these metals can be used in the annealed condition without built-up weld lands. Normally, most materials are in a heat-treated or cold worked condition to obtain high strength. Welding after heat treat tends to anneal the parent metal near the weld, thereby requiring a thicker section at the weld. Current technology fabrication costs with titanium alloys are high; therefore, it was not considered at this time. However, further effort to evolve an optimum pressure-fed system would have to include a study of the titanium alloys.

The weight estimates were based on the strength of 2014 T6 condition at room temperature. Strength of metals is higher at low temperatures; however, wall temperature during flight must be adequately determined and is not within the scope of this investigation.

V, A, Tankage Study (cont.)

3. Weight Considerations

Preliminary weight estimates were calculated neglecting local reinforcing at welds and discontinuities. Estimated weights are listed on the configurations, Figures V-1 through -5. For the designs that use plug volume, the required weight increase to convert this space to a pressure container was calculated to be 70,000 lb.

4. Conclusions

The configuration shown in Figure V-1 was selected as the representative system for evaluation. It consists of two spherical tanks joined by a stringer panel-rib intertank structure. The upper tank (oxidizer) has a wall thickness of 0.75 in. based on an ultimate factor of safety of 1.5. The lower tank (fuel) has a wall thickness of 1.3 in. This intertank structure serves to tie the two spherical tanks together and transmit the thrust to the upper stage. The additional structural weight needed to support the plug on a spherical tank is calculated to be 10,500 lb. Sphere reinforcement is accomplished by a ring located at the point of engine attachment to take out the discontinuity caused by thrust in the membrane shell. Thrust is then transmitted as a membrane load to the intersphere structure.

The weight of the cylindrical tankage shown in Figures V-4 and -5 is excessive. Use of cylindrical tankage as a pressure container is inefficient because the required wall thickness is twice that for a sphere of the same diameter and pressure; however, it does have the advantage that the tank can be used as a thrust transmission structure. For pressure levels associated with pressure fed systems, the weight penalty imposed by cylindrical pressure containers is not offset by using the same structure for thrust transmission.

V, A, Tankage Study (cont.)

Configurations using the plug volume do not appear to offer significant weight savings. These configurations were eliminated because of increased fabrication and their inherent complexity for an integral engine and tank design.

Although the configuration shown in Figure V-2 (single sphere with common internal bulkhead) is lighter than the standard configuration by approximately 5% and is shorter in length, system complexity appears to be greater. Some of the items that affect the complexity and reliability of the system are:

a. Interpropellant tank welds

b. Need for controlled differential pressure across the common bulkhead

c. Expansion and contraction of the sphere as it is subjected to different propellant temperatures.

V, Booster Design Studies (cont.)

B. THRUST-CHAMBER ASSEMBLY

1. Component Design Studies

a. Thrust-Chamber Assembly

The basic plug nozzle concept offers substantial performance and system integration advantages for a pressure-fed system. It requires the use of a throat in the form of an annulus, whose major radius is 40 ft, or more, while its width is only a few inches. The C-clamp construction required for this configuration requires very heavy structural members to maintain the inner and outer throat walls in their proper relationship. The high pressure volume is contained by two cylindrical walls of large radius and the structural weight of those walls is high. Heat transfer rate to the materials of the nozzle throat depends upon the minimum throat dimension so that the annular throat produces severe heat transfer problems.

The design adopted incorporates the advantages of the plug nozzle, while avoiding the difficulty of the annular throat. It consists of 32 cylindrical thrust-chamber subassemblies, the engine heat exchangers, an expansion cone, controls, and thrust take-out structure (see Figure V-6).

Thrust-chamber subassemblies (Figure V-7) incorporate a lightweight, sheet-metal injector and a film-cooled, stainless steel combustion chamber of a modified pen-shape design. The 32 subassemblies discharge upon a common truncated plug (33% isentropic length). Both helium and hydrogen heat exchangers are mounted in the upper part of the plug and are made of stainless steel tubing. The plug tip is of ablative material, with a honeycomb sandwich reinforcement.

V, B, Thrust-Chamber Assembly (cont.)

To keep the size of the individual chamber within reasonable limits and to avoid introducing excessive handling weight complications in the manifolding, while making maximum use of current experience and facilities, a thrust level for each chamber of approximately 200,000 lb is desirable. For convenient quadrant throttling, a number divisible by 4 is expedient; therefore, the number of units was fixed at 32 (thrust per chamber = 210,000 lb). The weight is not significantly affected by the number of chambers over the range 16 to 76.

The only practical means for cooling the walls of the thrust-chamber in the high pressure, high heat transfer region for this application appears to be the use of a cold fuel film. This method of cooling penalizes performance because it is not possible to burn all the propellant at optimum mixture ratio, but the alternatives have more formidable objections.

Use of reasonable costing ablative materials is, in the present state of knowledge, restricted to pressures below 750 psia plenum unless very severe throat erosion can be tolerated. The effect of throat erosion is to reduce the effective expansion ratio toward the end of the flight, at the time when high expansion ratios are most desirable. There exists very expensive refractories which give trivial erosion at 250 psia plenum pressure, but even if the expense were tolerable, fabrication methods for throats of the required size do not exist.

Regenerative cooling is, of course, fully feasible from the point of view of the thrust-chamber only. The added tank pressure required in the fuel tank (estimated to be at least 75 psi) requires a much heavier tank, and the performance penalty is intolerable.

V, B, Thrust-Chamber Assembly (cont.)

Radiation cooling has been successfully applied to bell nozzle skirts at sufficiently low pressure, provided the view factor for radiation to low temperature (400° R) approaches one-half. For this application, even for a heat sink at 20° R, the equilibrium temperature of the nozzle throat will exceed 3,000° F. Tungsten or molybdenum throats would be required, and the expense, weight, and fabrication difficulties are prohibitive.

Requirements for locating film cooling ports as well as the amount of film coolant to be injected were derived from this company's experience in the AJ 23-60 and XLR 113-AJ-1 programs.

Thrust-chamber performance parameters are delineated in Figure V-8.

b. Injector

Injector design is based on the XLR 87-AJ-5 unit. It is of two-piece construction, an oxidizer dome and a fuel manifold and orifice plate assembly. All parts are of formed sheet metal joined by continuous welding.

The low pressure drop available for injection requires ample manifolding and a fine grid injection pattern to obtain adequate atomization while expending minimum energy. The high-thrust-per-element concept is not attractive for this application because its pressure drop requirement is excessive.

V, B, Thrust-Chamber Assembly (cont.)

c. Ignition

Ignition will be accomplished by surface gap spark plugs. The efficiency and reliability of this system has been adequately demonstrated. Because components are light and cheap, four-fold redundancy will be used to further increase reliability. Four independent igniter circuits, each operating one spark plug per chamber, will be used. Each circuit will be capable of independently igniting all thrust-chamber subassemblies.

d. Expansion Plug

The contour of the external expansion plug is developed from the method of characteristics gas flow analysis. It is cut off at one-third of the ideal length. Although the resulting performance penalty is 1-1/2%, it provides an overall maximum in burnout velocity because of the concomittant weight reduction.

The high temperature is resisted by an ablative skin supported by honeycomb and a thrust take-out truss structure. Ablative cooling is required in spite of the high test costs resulting from the fact that the skin is not re-usable. Regenerative cooling requires excessive pressure drop, and radiation cooling is inadequate because of the limited view factor to a suitable heat sink.

Heat exchangers heating hydrogen and helium for pressurization are incorporated in the plug contour. Each is supported by a honeycomb backing and tied to the thrust support structure in such a way as to be independently serviced or removed without disturbing other components.

V, B, Thrust-Chamber Assembly (cont.)

2. Thrust-Chamber Assembly Controls

The thrust-chamber valves initiate propellant flow from the tanks during engine start and terminate propellant flow at the end of engine operation. They also throttle propellant flow, as required for thrust-vector control by chamber pressure modulation, during engine operation.

There are four fuel and four oxidizer valves for the engines assembly. One fuel valve and one oxidizer valve are mechanically linked. They control the flow to eight thrust chambers. Each pair is independently actuated by an electro-mechanical servo actuator. An electric motor drives two powdered-metal clutches, in opposite directions, in such a way that the desired output and direction depends only upon the particular clutch energized. Prior to start, the motor clutch assembly is started. Valve opening is initiated by energizing the opening clutch.

A study was made to determine the optimum valve design for minimum weight and pressure drop, and flow characteristics. Figure V-9 shows the weight-admittance characteristics of various valve designs. It can be seen that for K_w values less than 5,000, the gate slide valve is the lightest ($K_w = \frac{\dot{w}}{\left[(\ P)\ (\text{sp gr})\right]^{1/2}}$). All the various valve designs have approximately the same degree of complexity. The gate slide valve has a linear flow area-percentage open relationship, and therefore was selected. Because the system layout requires four "T" sections to connect the main propellant distribution lines to the respective quadrant manifolds, the gate slide valve was designed into the "T", which provided a weight saving. Figure V-10 shows this configuration.

V, B, Thrust-Chamber Assembly (cont.)

The dynamic conditions for both the fuel and oxidizer circuits were determined to obtain the appropriate valve size and weight. The following table summarizes results:

Circuit	Flow (lb/sec)	P (psi)	K_w	Valve Size Line Size (in inches)	Weight (lb)
Oxidizer	4,786	5	2,000	22	620
Fuel	798	5	1,310	19.5	370
Fuel	798	2.2	2,000	22	620

Because the fuel and oxidizer valves needed to satisfy a 5 psi design pressure drop were nearly the same size, 19.5 in. as opposed to 22-in., it was decided to make both the same diameter. In this way, the valves can be of identical design, which results in significant valve development and manufacturing cost savings. The pressure drop for the fuel circuit using a 22-in. valve was calculated to be 2.2 psi. A reduction in pressure loss on a pressure-fed system pays high dividends in over-all weight reduction and will more than correct for the increase in valve weight.

Figure V-10 shows the method of actuating the gate slide valve. The gate is attached to a ball-jack driven through a two-to-one gear train. Figure V-11 shows the required torque at the input to the gear train for valve actuation. This high starting torque results from the seal loading. Once the gate is removed from the seal, the torque becomes constant regardless of valve position. For two valves linked together, the opening stall torque for the actuator is 58 ft-lb. The steady-state torque is 24 ft-lb. For the maximum slucing speed, a shaft speed of 1,800 rpm is required. The steady-state power

V, B, Thrust-Chamber Assembly (cont.)

required for each bipropellant valve assembly is 10 hp. An electric motor magnetic clutch assembly was selected because it had the least complexity of any considered. Primary power will be battery-supplied.

Thrust-vector control by means of chamber pressure modulation is obtained by positional servo control of the respective valve assemblies at the four quadrants. Valve position feedback is obtained by attaching a transducer to the ball-jack shaft. This shaft is splined to prevent rotation during actuation. Figure V-12 indicates that flow through the valve is a linear function of valve position.

V, Booster Design Studies (cont.)

C. PRESSURIZATION SYSTEM STUDIES

The heart of the pressure-fed booster is the propellant pressurization system. Booster complexity is dictated to a large extent by this component system because it usually contains the majority of the booster functional components. Therefore, the total system reliability is highly dependent upon the complexity of the pressurization system. Hence, this system selection is of paramount importance in a pressure-fed-booster evaluation.

1. Pressurization System Requirements

If the booster is to operate at a constant performance level, the total feed pressure of the propellants must be maintained constant at the inlet to the thrust-chamber assembly throughout engine operation. Change in performance during operation is not necessarily undesirable, providing the performance change can be predicted and remains within allowable limits for satisfactory engine operation. Pressure at the thrust-chamber assembly inlet at any time during flight is governed by a combination of the static head of the propellant, the vehicle acceleration, and the propellant tank pressures. The total pressure resulting from static and acceleration heads throughout booster operation was determined for the standard vehicle, and is shown in Figure V-13. It is seen that if constant total pressure to the thrust-chamber assembly is to be maintained, oxidizer tank pressure must be somewhat decreased during the flight. If constant tank pressure is supplied, engine mixture ratio will rise as operation progresses.

At the beginning of the study program, exact system pressure drops were not available, thus tank pressure requirements could not be precisely determined. Initially, required oxidizer and fuel tank pressures were

V, C, Pressurization System Studies (cont.)

calculated to be 265 and 310 psia, respectively. The tank pressure requirements throughout engine operation were determined by integrating the static and acceleration head characteristics. They are summarized in Figure V-14. All pressurization systems considered were evaluated on this basis.

2. Systems Considered and Summary of Results

At the outset, many pressurization systems using various pressurants were analyzed for application to the subject booster. Many of these were eliminated immediately. Some showed adverse properties of the pressurant, incompatibility of the pressurant with booster propellants, inconvenience in using particular pressurants, poor logistics of pressurants, and very high weight or complexity of the total system. From this investigation feasible systems were selected for further analysis. They included the following:

a. Fuel Tank Pressurization

(1) Heated hydrogen pressurized by heated helium.

(2) Heated hydrogen pressurized by vaporized hydrogen.

(3) Exhaust gases from liquid bipropellant gas generator.

b. Oxygen Tank Pressurization

(1) Heated helium.

(2) Heated oxygen.

(3) Vapor self-pressurization (VāPak).

V, C, Pressurization System Studies (cont.)

(4) Exhaust gases from liquid bipropellant gas generator.

(5) Heated liquid nitrogen pressurized by heated helium.

No single pressurization system investigated was clearly superior to all other systems. All had some limitations, and final selection must be based on a relative evaluation of feasible systems. The liquid gas generator system for pressurization of either tank and the system using heated hydrogen pressurized by vaporized hydrogen for hydrogen tank pressurization appear to have very severe limitations, and further consideration of these systems is not warranted.

For oxidizer tank pressurization, the heated helium system is the lightest, thereby delivering the largest stage payload. However, the amount of helium required per launch is extremely large (25,100 lb) and represents 0.5% of the projected total yearly supply of helium in the United States. The heated oxygen system eliminates the need for helium, but imposes the weight penalty associated with the heavier oxygen gas. A further disadvantage, and probably more significant, is that the oxygen must be heated in a heat exchanger located on the engine cone. Any failures of such a heat exchanger would be catastrophic. Vapor self-pressurization (VaPak) offers the potentially simplest oxidizer system; however, system weight is excessive. If the oxygen pressurizing gas can be used to provide further thrust after liquid propellant exhaustion, the weight penalty is reduced and the system may become feasible. Use of liquid nitrogen pressurized by helium for oxidizer tank pressurization is also feasible, but is a heavy system and requires significantly more volume for storage of the required nitrogen and helium.

V, C, Pressurization System Studies (cont.)

Heated hydrogen pressurized by heated helium appears to be a satisfactory system for fuel tank pressurization. The other systems considered have serious limitations, as discussed previously, and are not recommended.

For the purpose of illustrating a complete booster system, and to facilitate a detailed weight and cost study of a representative system, heated helium and heated hydrogen were selected for oxidizer and fuel tank pressurization respectively. These systems are described in the next section along with an integrated analysis of the two systems. The other systems considered, together with their relative advantages and limitations, are discussed in subsequent sections. A comparative weight summary of the pressurization systems studied is given in Table V-1.

3. Pressurization System Selected for Detailed Study

a. Fuel Tank Pressurization - Heated Hydrogen Pressurized by Heated Helium

(1) System Description

Helium is stored at approximately 3,000 psia in a vessel located in the booster oxidizer tank. This helium is maintained at the temperature of liquid oxygen (-297°F). At a controlled pressure (504 psia), the helium is passed through a heat exchanger located in the engine nozzle plug and enters a liquid hydrogen pressurant tank at 565°R, pressurizing the tank to 398 psia. The liquid hydrogen, in turn, is forced through a heat exchanger in the plug, being vaporized and heated to approximately 465°R for use in pressurizing the liquid hydrogen fuel tank to 310 psia. A schematic diagram of the system is shown in Figure V-15.

V, C, Pressurization System Studies (cont.)

(2) System Advantages

(a) The pressurant for the fuel tank is the lightest possible and if condensed or absorbed cannot contaminate the fuel.

(b) Only a small volume of gas is stored under high pressure, providing low tank weights.

(c) Because large quantities of liquid hydrogen will be required for the booster fuel, the pressurant is available at the test and launching sites.

(d) The pressurants are stored as liquid or at extremely low temperatures, minimizing the size of containers.

(3) System Disadvantages

(a) Some helium is required, the logistics for which are poor. However, the amount required per launch (2,700 lb) is not considered a serious limitation for an application of this type.

(b) The amount of hydrogen pressurant required throughout engine operation may vary as a result of such factors as changing heat flux into the heat exchanger or propellant temperature change. Effects of these parameters on the over-all stability of the system are discussed in the integrated system analysis.

V, C, Pressurization System Studies (cont.)

b. Oxidizer Tank Pressurization - Heated Helium

(1) System Description

Helium, stored at high pressure (3,000 psi) in a vessel in the liquid oxygen tank, is reduced in pressure and passed through a heat exchanger in the engine nozzle plug, heating the helium to 565°R at an outlet pressure of 408 psia. A small part of the helium is bled off for use in pressurizing the fuel system and the remainder flows through a restrictor orifice to pressurize the oxidizer tank. This restrictor orifice reduces the pressure to the 265 psia required for the tank. The system is shown in Figure V-16.

(2) System Advantages

Heated helium offers the system of least weight for pressurization of the oxidizer tank. There are no problems of oxidizer contamination, or condensing or freezing of the pressurizing gas. When used in conjunction with the previously described "heated hydrogen--heated helium system" for the fuel tank, a common supply tank and regulator may be used, further simplifying the booster pressurization system.

(3) System Disadvantages

(a) The primary disadvantage associated with this system is the tremendous amount of helium required. For each launch, approximately 0.5% of the national yearly supply is expended. Combined with the helium required for development testing, the total amount required may eliminate this system from practical consideration.

V, C, Pressurization System Studies (cont.)

(b) The large quantity of helium needed for the oxidizer tank pressurization system amplifies the problems in development of a suitable gas-pressure regulator. Design problems for the regulator are not greatly increased, but the large quantities of stored, low-temperature, high-pressure gas required for testing greatly increase test facilities costs.

c. Analysis of Integrated Pressurization System

The two systems previously discussed were integrated to provide the booster pressurization system shown in Figure V-17. Helium for both systems is stored in a single tank. A burst diaphragm valve is used at the flow outlet of the helium tank to prevent helium leakage from the helium tank prior to booster operation. This valve is a squib-actuated butterfly type; it is equipped with a metal diaphragm that is sheared when the butterfly is operated. A secondary seal and closing squib is incorporated to provide reseal capability at engine shutdown, thereby terminating the helium supply to the pressurization system. This valve is shown in Figure V-18.

Helium pressure is reduced from the bottle storage pressure to system operating pressure by means of a pressure control valve which is located just downstream of the burst diaphragm valve. The valve is a ball valve design and has a maximum flow diameter of approximately 6 in. It is operated by a constant-speed electric motor that increases the flow area at a prescribed rate to maintain approximately constant tank pressure. The configuration of the ball flow part is designed to produce the required position-to-flow-area relationship. This valve does not have to seal because helium flow is initiated and terminated by the burst diaphragm valve. A sketch of the valve is shown in Figure V-19. This concept was used in preference to a conventional pressure regulator because of the large size and temperature environment of the valve. A pressure sensing device, acting through a torque motor, could be used

V, C, Pressurization System Studies (cont.)

to control the opening of the ball valve; however, accurate pressure measurement in a high-velocity gas line is difficult.

Burst diaphragms are incorporated upstream of the oxidizer and auxiliary hydrogen tanks to prevent oxygen and hydrogen from mixing, or entering the helium systems. These diaphragms are designed to rupture as the helium pressure rises during the start transient. They are reinforced on the helium side to prevent rupture from prepressurization of the propellant tanks.

The fuel tank pressurization bottle is located aft of the fuel tank and inside the plug. For weight estimating purposes, the material assumed was 2014 T6 at room temperature strength with a safety factor of 1.5 of ultimate allowable. Design parameters are indicated in the table after the following paragraph.

The helium pressurization tank is located inside the oxidizer tank. For weight estimating purposes, the material assumed was 2014 T6 at liquid oxygen temperature strength and with a safety factor of 1.5, based on ultimate stress. A weight saving of 31,000 lb could be realized by the use of 6 AL-4UA titanium alloy at liquid oxygen temperature. However, as shown in the Aerojet COSMOS study, the use of titanium alloys might be operationally incompatible with liquid oxygen. Therefore, for this investigation, titanium alloys were not selected.

V, C, Pressurization System Studies (cont.)

Design parameters are as follows:

Tank	Pressure (psi)	Dia. (in.)	Wall Thickness (in.)	Weight (lb)
Fuel Pressurization Tank	390	22.4	0.6	13,700
Helium Tank	3,000	21.9	3.67	78,800

A preliminary analysis of the integrated pressurization system was made to determine the overall system characteristics and problem areas requiring further evaluation. The anlysis revealed the following:

(1) Temperature of the pressurizing gases in the propellant tanks throughout a firing is difficult to predict by theoretical analysis, and currently available test data is insufficient for determining these temperatures.

For purposes of this investigation it was assumed that twice the calculated quantity of pressurizing gases (assuming no cooling) will be necessitated by the fact that the pressurizing gases are cooled in the propellant tanks. This assumption is believed to be conservative, particularly for vessels of the size being considered.

(2) Analysis of the system has shown that, because of the density of the residual helium in the supply vessel at the end of a firing, a significantly smaller amount of helium is required if it is stored at liquid oxygen temperature (-297°F) than at liquid hydrogen temperature (-421°F) (25,100 lb vs 36,067 lb). Obviously, there is an optimum storage temperature other than these, but storage in the liquid oxygen tank or the liquid hydrogen tank offers the only practical means for maintaining a constant helium supply temperature. Methods for heating the helium in the supply vessel during a firing should be investigated further.

V, C, Pressurization System Studies (cont.)

(3) A special study was conducted to determine the heat exchanger requirements for heating the pressurants for tank pressurization. The heat exchanger analysis is presented in section V, C, 5, of this report.

(4) The pressurization system design is greatly simplified if a constant mass flow is supplied to the nozzle plug heat exchanger. For this condition to exist, it is necessary that the helium be supplied to the exchanger at constant pressure and temperature. Constant pressure is maintained by the pressure regulator. To compensate for temperature change resulting from expansion of the helium in the supply tank, the helium is passed through a heat exchanger submerged in the oxidizer tank. This maintains a constant temperature at the inlet to the helium pressure regulator. The helium temperature drop is negligible as it passes through the regulator because the gas temperature and pressure are near the inversion point of helium (Joules - Thompson effect).

(5) It may be advantageous to separate the helium supply vessels so that the large volume required for the oxidizer system can be used for a lower end-pressure (oxidizer tank pressure is 265 psia as compared with a fuel tank pressure of 310 psia). In this way, the required helium is reduced from 25,100 to 23,344 lb. Another advantage realized is that the burst diaphragms at the inlets of the oxidizer and auxiliary fuel tanks can be eleiminated because the oxidizer and fuel vapors are naturally isolated. Disadvantages of the system are the need for two helium pressure regulators and two helium tanks.

V, C, Pressurization System Studies (cont.)

(6) In the initial analysis of the systems, it was necessary to assume that conditions were essentially constant in various parts of the systems. Typical of these assumption are: constant temperature in the main propellant tanks; constant propellant flow rates; no effect of missile acceleration; and constant temperature and density conditions in the heat exchangers.

In some cases, the effect of variations in these conditions was recognized and corrected by applying necessary correction factors. In other instances, exact conditions could not be determined, or they involved lengthy and complex computer programs or experiments. Without full knowledge of these characteristics, it is possible that the depicted system may be unstable (i.e., fail to recover from certain transient conditions that could be encountered during operation).

The system is designed for a constant volume flow rate of pressurizing gas into the propellant tanks. Flow rate is governed by the pressure differential between the regulator and the propellant tanks as well as by the conditions in the heat exchanger. Although possibly exaggerating the conditions that could occur, the following illustrates the need for more exacting analysis of system operation during transient conditions.

(a) An increase in the pressurization gas requirement could be caused by the sudden cooling of gas in the liquid hydrogen tank (from possible propellant sloshing), lowering pressure in this tank. The flow of pressurant through the system is increased because of the increase in differential pressure; however, the maximum gas flow rate is fixed by system pressure drop and the increased gas flow may not be adequate to compensate for the pressure decrease in the propellant tank before unstable thrust chamber operation occurs

V, C, Pressurization System Studies (cont.)

as a result of the insufficient propellant flow. Also, the increased mass flow through the heat exchangers causes gas at a lower temperature to be produced, thereby aggravating the situation. It is conceivable that a continued decline in tank pressure could result.

(b) A reduction in requirements for pressurization gas could result in overheating the gas in the heat exchanger, which, in turn, would further reduce the flow requirement. This process could progress until failure of the heat exchanger resulted.

(c) Providing directional control by throttling propellants in certain sections of the thrust-chamber has been considered. This change in propellant flow rate may create a control problem for the pressurization heat exchanger system.

(7) After studying the stability aspects of the designed pressurization system, an alternative one was designed. The latter appears to be inherently more stable and is shown in Figure V-20. In this system, the oxidizer and fuel pressurization systems are completely isolated, thereby preventing any feedback from one to the other.

In each system, helium--at high pressure--is stored in separate tanks within the oxidizer tank. This helium is passed through a heat exchanger and reduced in pressure by means of a pressure regulator or programed flow-area regulator to maintain approximately constant density as it flows into the plug heat exchangers. The heated helium in the oxidizer system is passed through an additional pressure regulator, which further reduces the pressure before it enters the oxidizer tank. In the fuel system, the heated helium is used to pressurize the auxiliary hydrogen tank, and hydrogen from this tank is passed through a heat exchanger and then regulated into the main hydrogen tank.

V, C, Pressurization System Studies (cont.)

4. Other Systems Considered

a. Fuel Tank Pressurization System

(1) Heated Hydrogen Pressurized by Heated Vaporized Hydrogen

(a) System Description

Liquid hydrogen from an auxiliary storage bottle is heated in a heat exchanger located on the engine cone and is used to pressurize the main fuel tank. The heated gaseous hydrogen is passed back through another heat exchanger located in the auxiliary tank to supply heat to the residual hydrogen for pressure generation so that flow can be maintained through the cone heat exchanger. The system is shown in Figure V-21. A layout showing this system in conjunction with a heated oxygen system for oxidizer tank pressurization is shown in Figure V-22.

(b) System Analysis

This system is somewhat similar to the selected system, with the main difference being the means of providing the pressure head in the auxiliary tank. Elimination of helium represents a significant advantage if the required pressure head can satisfactorily be obtained by heating the hydrogen pressurant.

V, C, Pressurization System Studies (cont.)

To obtain the required driving pressure of about 350 psia in the auxiliary hydrogen tank, the auxiliary and main tanks must be originally pressurized with helium to initiate flow when desired. As flow from the tanks begins, hydrogen from the auxiliary tank is vaporized by the heat exchanger in the engine cone. The resulting warm gaseous hydrogen returns to the main tank via the heat exchanger located in the auxiliary tank. As heat is transferred to the hydrogen in the auxiliary tank, it expands and is dispelled, thereby completing the loop. Because the auxiliary hydrogen would continuously decrease in density (and increase in temperature), the volumetric flow rate would have to continuously increase to supply a constant weight flow rate of gaseous hydrogen to the main heat exchanger. To obtain the proper varying volumetric flow rate from the auxiliary tank, heat input to the auxiliary hydrogen would also have to continuously change. Proper control of a varying heat input would be very difficult and would be greatly complicated by the changing heat transfer characteristics of hydrogen on both sides of the auxiliary heat exchanger, the changing temperature, and by the changing useful heat transfer area.

It was concluded that, even if this system is theoretically possible, the control system required would be very complicated. Also, development problems would be of such magnitude as to create more disadvantages than the advantages to be gained by eliminating helium.

(2) Exhaust Gases from Liquid Bipropellant Gas Generator

(a) System Description

Pressurization gases are obtained from a fuel-rich gas generator that burns hydrogen and oxygen pumped from the main propellant tanks. Turbopump power is provided by a single turbine operated by a small

V, C, Pressurization System Studies (cont.)

portion of the generator exhaust gases. The exhaust gases are cooled by heat transfer from the main engine propellant to condense the water vapor. They are then used to pressurize the propellant tank. This system is shown in Figure V-23.

(b) System Analysis

The use of a liquid bipropellant gas generator provides a relatively high temperature, low molecular weight gas for fuel tank pressurization. The gas generator system considered burned a fuel-rich mixture of hydrogen and oxygen, operating at a mixture ratio of 0.5. The combustion temperature at this mixture ratio is 500° F. Combustion products are 62% hydrogen gas and 38% water vapor, by weight.

It was assumed that twice the calculated quantity of gas will be required because the pressurizing gases will be cooled in the liquid hydrogen tank. It was also assumed that the gas must be free of water vapor to prevent ice formation in the hydrogen tank. The temperature of the exhaust gas must be reduced below 400°F at the required 310 psi before the vapor can be condensed.

Flow rate calculations were based on the above assumptions. Oxygen and hydrogen flow required are 49 and 99 lb/sec, respectively. Oxidizer and fuel pumping horsepower requirements to provide these weight flows are 48 and 1,180, respectively.

V, C, Pressurization System Studies (cont.)

The turbomachinery required to pump the propellants is in the size class of the main Titan second-stage turbopump. If the potential reliability of any well designed turbopump of this size--or larger--is assumed to be the same, then any advantage claimed by a pressure-fed system in comparison to a similar pump-fed system on the basis of elimination of the turbopump from the main propellant feed system is lost. Therefore, it was decided that the gas generator pressurization method will not afford the system simplicity desired in pressure-fed boosters.

The requirement to condense and remove the water vapor at the rate of 56 lb per sec would also present a complex design configuration, and probably the weight of the heat exchanger required for this purpose would nullify the comparative lightness of the rest of the system. Because the system looks unattractive from the turbopump requirement alone, design of the water condensation equipment was not undertaken.

b. Oxidizer Tank Pressurization System

(1) Heated Oxygen

(a) System Description

Oxygen from the main oxidizer tank is heated in a heat exchanger and then is used to pressurize the oxidizer tank. The total oxidizer propellant head forces the oxygen to flow through the heat exchanger. Constant inlet conditions to the heat exchanger are maintained by means of a pressure regulator. This system is illustrated in Figure V-24. A system layout is shown in Figure V-22.

V, C, Pressurization System Studies (cont.)

(b) System Analysis

A heat exchanger to produce oxygen gas from liquid oxygen is subject to rather severe design limitations. The combustion gas produced by the thrust chamber (total temperature above 5,000°F) is the only available source providing sufficient heat. Because the wall temperature of the heat exchanger must be kept well below the melting point of the metal, the heat transfer coefficient on the cold side of the wall must be kept high. As long as there is no phase change in the liquid oxygen, an adequate cold-side heat transfer coefficient can be maintained; but when boiling starts, large local reductions in heat transfer coefficients occur. The mechanism believed to be involved is the formation of gas bubbles at wall irregularities. A bubble, unless it is immediately swept away, insulates the cold side of the wall, the wall temperature rises to the melting point of the metal, and the wall fails.

The consequences of a failure are catastrophic. At--or somewhat below--their melting points*, steel, aluminum, copper, nickel, monel, titanium, and other metals ignite and burn vigorously in the presence of oxygen as well as in water vapor, CO_2, and other highly oxidized atmospheres. Combustion of the metal normally continues until either metal or oxidizer is exhausted, or until a change of section provides an adequate heat sink to cool the surface well below the melting point.

---

* Dean, L. E. and Thompson, W. R., AGC Report No. LRP-150 (Special), Ignition Characteristics of Metals and Alloys, 31 December 1959.

V, C, Pressurization System Studies (cont.)

To avoid the possibility of catastrophic failure, it is necessary to provide walls that maintain structural soundness in contact with gaseous oxygen at 5,000°F, or to prevent phase change within the heat exchanger. Suitable materials are lacking for the first alternative. The second requires that the heat exchanger be operated either above or below the critical pressure, but above the desired tank pressure.

When operating at pressure above the critical pressure of oxygen (730 psia), an operating pressure of about 1,000 psia will be required to prevent local velocity effects from producing local static pressures below the critical.

The heat exchanger can also be operated below the critical pressure, but above the desired tank pressure. For example, if tank pressure is 350 psia and the heat exchanger operates at 500 psia, the liquid in the heat exchanger is heated to the saturation temperature at 450 psia, leaving a margin of 50 psia to suppress boiling. When the superheated liquid is introduced into the tank through a throttling valve, a portion of it (approximately 10% less than the assumption taken) flashes into vapor. This system requires ten times the volume in the auxiliary oxidizer tank as does the vapor critical system, and the weight penalty is even greater.

(2) Exhaust Gases from Liquid Bipropellant Gas Generator

(a) System Description

Pressurization gases are obtained from an oxidizer-rich gas generator burning hydrogen and oxygen pumped from the main propellant tanks. Turbopump power is provided by a single turbine operated by

V, C, Pressurization System Studies (cont.)

a small proportion of the generated exhaust gases. These exhaust gases are cooled to condense the water vapor and are then used to pressurize the propellant tank. This system is shown in Figure V-25.

(b) System Analysis

This system represents the counterpart to the gas generator system for oxidizer tank pressurization. The gas generator operates at a mixture ratio of 94, at which the combustion temperature is 1,410°F. Ten percent, by weight, of the exhaust products is water vapor that must be condensed out.

Required oxidizer and fuel propellant flow to provide the necessary exhaust gases were calculated to be 508 and 5.4 lb/sec, respectively. The pump power required to supply these flow rates at their respective head rises was computed to be 500 hp for the oxidizer and 64 hp for the fuel. This oxidizer pump is similar in size to the Titan second-stage turbopump (791 hp).

The same limitations found in the fuel tank gas generator pressurization system are present in this system; namely, the large turbopump requirement and the water condensation problem. Therefore, evaluating on the same basis, the system is considered undesirable.

V, C, Pressurization System Studies (cont.)

(3) Vapor Self-Pressurization (VāPak)

(a) System Description and Operation

Tank pressurization is obtained from the vapor pressure of the propellant in the tank. Prior to launch, propellant in the tank is maintained at the temperature necessary to provide a vapor pressure equal to the desired tank pressure. When the tank pressure is lowered by propellant expulsion, the vapor-liquid equilibrium is upset and liquid vaporizes, creating additional gas thereby tending to counteract the decrease in pressure. The total decrease in pressure at propellant exhaustion is approximately 30% of the initial pressure. A layout for this booster system, using VäPak for oxidizer tank pressurization, is shown in Figure V-26.

(b) System Analysis

In any closed container, partially filled with a liquid, vapor is present at a definite pressure. Under equilibrium conditions, this vapor pressure is a function of the nature of the liquid and its temperature only. When the temperature of the liquid is reduced by the vaporization as a result of propellant expulsion, the equilibrium vapor pressure is also lowered. Liquid heat of vaporization is supplied from that stored in the body of the liquid. The magnitude of the decrease in pressure is a function of the proportion of liquid expelled, specific heat of the liquid, its latent heat of vaporization, specific volume of the liquid, specific volume of the vapor, and the vapor-pressure temperature relationship.

V, C, Pressurization System Studies (cont.)

This principle of tank pressurization has been applied by Aerojet-General to propellant expulsion with several different propellants. Tests with liquid oxygen have shown that the final tank pressure is approximately 70% of the initial pressure.

Vapor pressurization for this application provides a relatively simple pressurization system. The only control hardware required on the booster would be a pressure relief valve to maintain the desired pressure prior to launch. Assuming an initial tank pressure of 265 psi and a 30% pressure loss during booster operation, tank pressure at engine burnout is 186 psi. Because of the booster acceleration, tank pressure required at this time to maintain constant engine mixture ratio has dropped from 265 to 217, thus tending to counteract the loss of tank pressure. The net unbalance gives a slight decrease in mixture ratio, which is not considered objectionable.

The limitations of the VāPak system are largely the weight penalties associated with it. Oxygen must be in vapor-liquid equilibrium at all times, and its initial temperature must correspond to the vapor pressure desired for tank pressurization. The temperature required to obtain 265 psi, is -228°F, at which point the liquid density is 55.7 lb/cu ft. To contain the required mass of propellant, a propellant tank volume must be increased 26.5% above that required for oxygen stored at its normal boiling temperature (ρ = 70.4 lb/cu ft). Because the weight-to-volume relationship of spherical tanks is approximately proportional, the oxidizer tank weight is also increased by 26.5%, or approximately 19,400 lb. The properties of saturated oxygen are summarized in Figure V-27.

V, C, Pressurization System Studies (cont.)

At propellant exhaustion, the tank is filled with saturated oxygen vapor at 186 psia. The corresponding density is 3.2 lb/cu ft. Based on a tank volume of 46,600 cu ft, 149,000 lb of oxygen remains in the tank. Unless this gas can be burned to provide further thrust, the entire gas weight is charged to the system inert weight. A special study was conducted of this subject, and is reported in section V, C, 5, of this report.

(4) Heated Nitrogen Pressurized By Heated Helium

(a) System Description

Cold helium is stored at high pressure. It is reduced in pressure, warmed by passing through a heat exchanger in the nozzle plug, and flows into a tank of liquid nitrogen. This nitrogen is forced through another heat exchanger in the nozzle plug, where it is vaporized and heated to pressurize the oxidizer tank. It is the same as the heated hydrogen-heated helium system described for the dual tank, except for the temperatures, pressures, and the use of liquid nitrogen to replace liquid hydrogen. The system is shown in Figure V-28..

(b) System Analysis

The most significant feature of this system is the gross reduction in the quantity of helium required. When used in conjunction with the heated hydrogen system for fuel tank pressurization, helium requirements are reduced from 25,100 to 6,314 lb. The use of nitrogen does not introduce problems of compatibility between the pressurization gases and the oxidizer, nor does it greatly increase system complexity.

V, C, Pressurization System Studies (cont.)

The density and other properties of nitrogen are more favorable for this application than those of any other gas that can be used, excepting helium. However, in using liquid nitrogen to replace helium, a system weight penalty of approximately 51,000 lb is incurred.

Because liquid nitrogen can be manufactured in large quantities on site, development costs would be reduced considerably.

The problems associated with the use of liquid nitrogen are identical with those of the hydrogen-helium system, except for weight and locating the liquid nitrogen vessel in the booster. If the liquid nitrogen were stored in a spherical vessel, a tank 16.25 ft. in diameter is required. Such a system would probably be packaged by using the plug volume to store the liquid nitrogen sphere and relocating the auxiliary hydrogen tank for fuel tank pressurization to inside the main fuel tank.

5. Special Pressurization System Studies

a. Heat Exchanger Requirements

A special study was made to evaluate the feasibility of heating helium and hydrogen for propellant tank pressurization, as used in the pressurization system discussed in Section V, C, 3. If found feasible, pressurant temperature obtainable, amount of pressurants required, and heat exchanger design data would be determined.

V, C, Pressurization System Studies (cont.)

The following assumptions were made for this study:

Because the pressurizing gases are cooled in the propellant tanks, it was assumed that twice the calculated quantity (assuming no cooling) of gas will be required.

Allowable wall temperature of all heat exchangers was limited to 1,460° R.

For plug heat exchangers, adiabatic wall temperatures were taken as 5,620° R.

Required propellant tank volumetric flow rates are as follows:

Oxidizer: 279 cu ft/sec at 265 psia

Fuel: 778.9 cu ft/sec at 310 psia

Heat from the engine exhaust gases is the only existing source for heating pressurization gases; therefore primary investigation effort was devoted to analyzing heat exchangers located on the engine plug.

Heat transfer coefficients were evaluated by two methods: expressions for aerodynamic heating; and expressions for heat transfer in annular areas. Results were comparable; however, aerodynamic heat transfer coefficients were more conservative and were used as the basis for evaluating heat exchanger requirements.

V, C, Pressurization System Studies (cont.)

The plug heat transfer coefficients as functions of X (x), (Figure V-17) were as follows:

X (x) (feet)	Plug Radius (in.)	h (Btu/hr-sq ft-°F)
1.639	262	359
2.738	252	289
4.256	238	230
6.358	222	184
9.268	203	147
13.325	181	105

Because the pressurants represent vehicle inert weight, it is desirable that they be heated to as high a temperature as possible. For maximum velocity, and consequently the highest heat transfer coefficient per given weight of fluid, the tube size was taken as the smallest size allowable by design limitations. This was set at 3/8-in. steel tubing with 0.020-in. wall thickness.

Tube-side heat transfer coefficients for both helium and hydrogen were evaluated from the expression

$$h_c = .0217 \frac{k}{D_e} R_e^{0.8} Pr^{0.4} \left(\frac{T_B}{T_w}\right)^{0.37}$$

where:

D_e = equivalent tube diameter

R_e = tube Reynolds number

T_B = fluid bulk temperature

T_w = wall temperature

V, C, Pressurization System Studies (cont.)

Steady-state flow and heat transfer only were considered. That is, at any section in the heat exchanger the expression

$$hg\,(Tg - T_w) = h_c\,(T_w - T_c)$$

was assumed to describe operating conditions. For this study, wall temperature gradients were neglected. Maximum attainable pressurant temperatures were then established.

Plug heat exchanger requirements are:

Pressurant	Weight Flow Per Sec Per Tube	Inlet Temp (°R)	Outlet Temp (°R)	p (psi)	Wall Temp (°R)	No. of Tubes
Hydrogen	0.0910	40	465	50	1,400	2,144
Helium	0.1851	40	565	100	1,400	640

In the region discussed, the plug contains 4,410 3/8-in. tubes along its periphery.

It is emphasized that these conditions are for the length of tube stated, which at the outlet was arbitrary. If it is later determined that all of these tubes on the periphery are not required, the length may be changed and tube weight flows adjusted to cover the total periphery of a region. It can be assumed that the same coolant temperature will be obtained; therefore, the pressurant requirements will remain unchanged.

V, C, Pressurization System Studies (cont.)

Assuming that the above temperatures are attainable in the plug heat exchanger, the required weight flow of pressurants are found to be as follows:

Tank	Pressurant	*Vol Pressurant Required (cu ft/sec)	Gas Flow
Fuel	Hydrogen	1,557.8	195 lb/sec at 465°R and 310 psi
Auxiliary hydrogen	Helium	89.30	20.64 lb/sec at 565°R and 350 psi
Oxidizer	Helium	557.8	97.61 lb/sec at 565°R and 265 psi

The number of heat exchanger tubes occupied are, then, $\frac{195}{0.0910}$ (or 2,143 hydrogen tubes) and $\frac{118.25}{0.1851}$ (or 639 helium tubes), or a total of 2,782 of the 4,410 available tubes.

The helium stored in the oxidizer tank is initially at 3,000 psia and 163°R (liquid oxygen temperature). During the firing, the helium pressure drops, and it is assumed that the residual helium follows an isentropic expansion; the helium temperature will drop to 104°R (at 500 psi). A decrease of 66% in helium supply density from 5.3 lb/cu ft to 1.8 lb/cu ft results. This drop in temperature is undesirable because it is best to maintain a constant weight flow of helium throughout the run. In order for a pressure regulator to supply constant weight flow at a given outlet pressure, the temperature of the fluid being admitted must be constant. Thus, it was decided to incorporate a heat exchanger into the oxidizer tank to maintain a constant inlet helium temperature to the regulator at 163°R $\pm$ 2°R.

* Based on Volumetric Pressurization Factor = 2

V, C, Pressurization System Studies (cont.)

Heat exchanger requirements were based on conditions at the end of the run, at which time it will be necessary to increase the temperature of 118.4 lb of helium per second from 104° R to 161° R.

The heat exchanger consists of tubing arranged in the lower part of the liquid oxygen tank. This tubing will be immersed in liquid oxygen for longest time possible so that the oxygen flowing out the bottom will warm the colder helium flowing within the tubes.

The heat exchanges was assumed to be made of 3/8-in. tubes, spaced at the vertexes of equilateral triangles, with 5/8-in. center-to-center distance. Pressure drop limitations in the heat exchanger required that the tube lengths be short and that each transport a small amount of fluid. It was determined that 1,800 3/8-in. tubes, each 10.5-ft long meet both the temperature requirements and the pressure drop limitations. Total length of tubing is 189,000 ft with weight of 1,411 lb (based on 0.020-in. wall). The heat exchanger can be fitted into the lower 2.9 ft of the liquid oxygen tank so that it is immersed until during the last second of firing.

The residual helium at the end of the run is 8,035 lb, based on 500 psi tank pressure. When the total system is integrated, pressure losses and further limitations will cause a variance in this figure.

V, C, Pressurization System Studies (cont.)

As a result of the heat exchanger studies, the required propellant and pressurant tank sizes were determined to be as follows:

Vessel	Volume (cu ft)	Diameter (ft)
Fuel tank	102,815	58.13
Auxiliary hydrogen tank	5,894	22.41
Helium tank	4,465	20.43
Oxidizer tank	41,280	42.88

b. Engine Operation on Residual Tank Pressurization Gases (Pressure-Fed Systems)

A preliminary study to determine the feasibility of using residual tank pressurization gases for engine operation was performed. It is estimated that a total impulse of 34,000,000 lb/sec over a 20-sec period can be obtained for this vehicle system. Thrust at the end of 20 sec is 600,000 lb. Considering the greatly reduced mass of the vehicle and its velocity at the time gaseous operation begins, the potential to continue at a reduced thrust appears attractive; but a more comprehensive study should be made. The time required for complete depletion of gases from the tanks was not determined in this study.

The assumptions made for this study follow:

The injectors will function properly with gaseous flow. (Other companies report no difficulty in traversing gaseous transient flow conditions.)

V, C, Pressurization System Studies (cont.)

Temperature within the tanks remains constant as pressure decreases. Only vapor density is affected. (A weight flow of 800 lb of H_2 results in a pressure drop of only 19 psi and is accompanied by a temperature decrease of approximately 0.2°F.) This does not seem too serious because the tank walls contain sufficient hast to offset any sudden decrease in temperature.

Diffusion of helium into the main hydrogen tank is not great enough to affect the combustion and therefore, not taken into consideration when calculating mixture ratio. (Unless the helium within the hydrogen pressurization bottle is isolated, its diffusion, which is rapid, into the main propellant tank can greatly nullify any beneficial use of tank pressurization bases).

Film cooling will be ineffective and chambers will have to be replaced after each flight. (Expected temperature at M.R. = 10 is approximately 5,000°F.)

Propellant feed line resistances are taken at 10 psi and do not change with decreasing pressure. (When the system is finalized, these line resistances must be ascertained to a fairly close approximation in order to accurately determine weight flows and mixture ratios.)

The weight flows from the tanks were calculated from the general gas equation.

$$PV = \mathcal{N} RT = \frac{W}{M} RT$$

$W = \frac{PVM}{RT}$, which is the total weight of gas available in each tank. A change in the total weight per unit of time is the weight flow to the injector.

$$\dot{w} = \frac{dW}{dt} = \frac{\partial W}{\partial P} : \frac{dP}{dt} + \frac{\partial W}{\partial T} \cdot \frac{dT}{t}$$

V, C, Pressurization System Studies (cont.)

Since temperature is not considered, the weight flow from the tanks may be simply expressed as

$$\dot{w} = \frac{VM}{RT} \cdot \frac{dP}{dt}$$

The weight flow through the injector orifices is expressed as follows

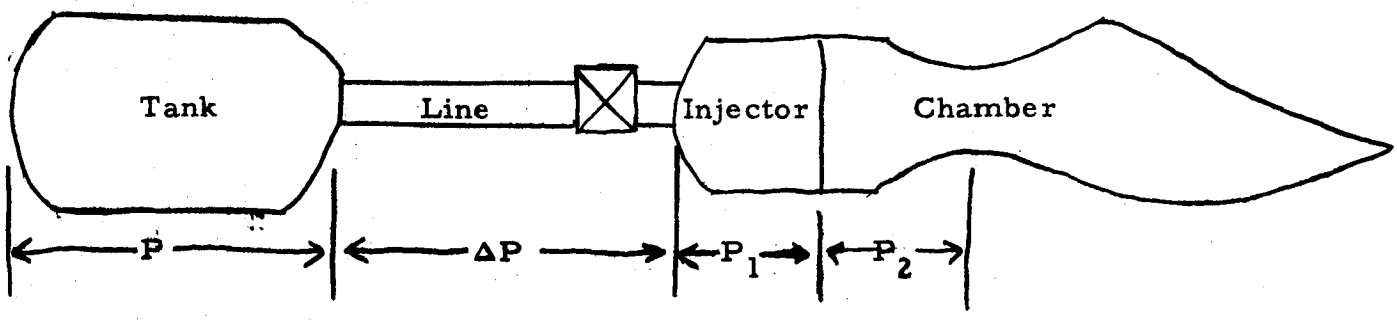
$$\dot{w} = A\sqrt{\left(\frac{2kg}{K-1}\right)\left(\frac{P1}{v_1}\right)\left[\left(\frac{P2}{P_1}\right)^{2/k} - \left(\frac{P2}{P_1}\right)^{\frac{K+1}{K}}\right]}$$

Since the weight flow from the tanks equals the weight flow to the injectors, the equations may be set equal.

$$\frac{VM}{RT} \cdot \frac{dP}{dt} = A\sqrt{\left(\frac{2kg}{K-1}\right)\left(\frac{P1}{v_1}\right)\left[\left(\frac{P2}{P_1}\right)^{2/k} - \left(\frac{P2}{P_1}\right)^{\frac{K+1}{K}}\right]}$$

P_1 is the injector pressure and P_2 is the chamber pressure. The injector pressure is equal to the tank pressure minus the resistance of the propellant feed lines between the two systems. This was taken as 10 psi in all calculations and should be reconsidered when the design is finalized.

The system studied is shown below:



V, C, Pressurization System Studies (cont.)

The pressure drop through the lines to the injector may be determined by using the fanning equation.

$$P = \rho_f \frac{LV^2}{2gD}$$

The specific volume was determined for each new injector pressure as follows

$$v_1 = \frac{P_1}{RT}$$

The initial chamber pressure chosen was taken at the point where critical pressure across the orifices is first obtained.

$$P_2 = P_1 \left(\frac{2}{K+1}\right) \frac{K}{K-1}$$

The ratio of the weight flows is calculated for each second of operation. A corresponding value of the characteristic exhaust velocity is then obtained. (The values used in this study were only rough approximations.) The chamber pressure is then calculated by using the following equation

$$Pc = \frac{c^* W_t}{A_{tg}}$$

V, C, Pressurization System Studies (cont.)

Values for the thrust coefficient were not available and therefore an average 1.7 for vacuum condition was used for all calculations. The thrust was then calculated by the usual method

$$F = C_f \, Pc \, A_t$$

A total impulse was then obtained for the first 20 sec of operation from the equation

$$I_t = \int_0^{20} F dt$$

The results are shown in Figure V-29.

V, Booster Design Studies (cont.)

D. VECTOR AND ROLL CONTROL

Vector control of plug nozzle engines can be accomplished by chamber pressure modulation, secondary injection, or by using auxiliary thrust chambers. The most simple and attractive method is by chamber pressure modulation--if the system is capable of providing the required control moment.

A study was made to determine the suitability of chamber pressure modulation for this application. The engine thrust chambers were segregated into quadrants of eight chambers each, and propellants to each quadrant were controlled by a single set of linked oxidizer and fuel valves. Two opposed quadrants were used for pitch and yaw control, respectively.

The available control moment was calculated* for thrust-vector control obtained by chamber pressure modulation. The analysis yields two, usually opposed, moments. The larger, $M_{//}$, is produced by the lateral shifting of the main thrust-vector toward the sector with the higher pressure. The smaller, $M_{\perp}$, is the result of tilting the thrust-vector toward the sector with the lower pressure. Table V-2 shows these moments and their albegraic sum, M, as a function of pressure difference in two opposed chambers. The last column of the table shows the moment obtained from a pair of opposed eight-chamber (90°) sectors of pressure difference ΔP.

* AGC Report No. LRP 210, Thrust-Vector Control for Plug Nozzles, 8 March 1961

V, D, Vector and Roll Control (cont.)

Because the thrust-vector control moment is estimated at 12,500** ft kips, and experience shows that it is undesirable to reduce the flow below approximately 60% of the nominal to avoid serious combustion stability problems, some increase in control moment as a function of pressure difference is required. This increase can be obtained by controlling a 180° sector (at the expense of considerable complication in the sensing and guidance system loops) or by changing the nozzle contour to increase the available moment. Small performance penalties accompany the increased moment obtained by nozzle modification. A detailed study is required to determine the optimum relationship between control moment and specific impulse. Figure V-2 shows the magnitude of this relationship. The moments and specific impulse losses shown were obtained by changing the attitude (angle of throat plane to missile axis) only. Thus, specific impulse losses are overestimates for the control moments shown.

It was concluded that vector control by chamber pressure modulation is practical for this booster configuration, at the expense of a small performance loss. Power requirements to operate the throttling valves were determined, and are given in the thrust-chamber valve discussion, Section V, B, 2, d of this report.

** An appreciable contribution to the required control moment, as estimated, comes from assumed manufacturing inaccuracies which cause the thrust-vector in the no-control condition to be displaced from the geometric axis. A detailed study of the manufacturing tolerance problem may reveal economical means for reducing the displacement at no-control, and hence the over-all control moment required. The control moment requirement may be alternatively reduced by modifying the vehicle configuration to shift the center-of-gravity forward or the center-of-pressure aft. The optimum relationship between vehicle drag and engine Isp can be determined only by model testing.

V, D, Vector and Roll Control (cont.)

A study to determine the roll control requirements was not within the scope of this study. However, roll control provisions are included in the booster configuration and consist of two auxiliary engines located 180° apart on the booster major diameter. The required control moment and thrust of these engines was taken to be approximately the same as those used in the Cosmos study (an evaluation of a 6-million-lb booster).

V, Booster Design Studies (cont.)

E. PROPELLANT TANKAGE LOADING SYSTEM

1. Hydrogen Tank

a. Fill System

Assuming a fill time of 4 hr with a use factor of 1.5 and a tank volume of 767,363 gallons, the fill rate for the hydrogen tank is 5,000 gpm.

Prior to fill, the tank is chilled with liquid nitrogen and purged with gaseous hydrogen. Purging is accomplished by supplying the hydrogen gas through the tank vent system and allowing the tank to bleed through the fill and topping connections in the tank discharge lines. In this way, the dense nitrogen gas will be removed from the tank.

The tank is filled through an 8-in. manually disconnected valve-coupling mounted on one thrust-chamber valve. Mass flow meters in the fill and vent lines or tank level sensors are used to determine the tank fluid level. When the tank is full, the fill coupling is disconnected and the tank level is maintained by topping through four 1-in. couplings, one on each thrust-chamber valve. The mass flow meters or tank level sensors are used as a means of maintaining constant propellant level during countdown or hold-fire periods. Figure V-31 is a design of a manually connected fill or vent valve-coupling, which requires no springs for proper action. The coupling can be disconnected only after the booster has been sealed. A ground-supplied actuator may be used to disconnect as a safety precaution. The topping coupling is automatically disconnected by a lanyard before the firing.

V, E, Propellant Tankage Loading System (cont.)

Because the tank discharge lines outlets are at a higher level than the inlets, a gas lock may develop during the fill cycle. If this becomes a problem, the lines will be bled through the topping couplings during the fill cycle until the tank fluid level is above the line outlet level and the fluid is thoroughly chilled.

During the fill cycle, the tank is vented through an 18-in. (characteristic diameter) coupling of a design similar to the fill coupling. A separate combination topping vent and pressure relief valve (see Figure V-33) is used for gas-off during the topping cycle. The vent-relief valve area is large enough to preclude choking at the topping rate of 325 gpm.

A study was made to determine the tank fill rates with varying degrees of tank insulation. Figure V-33 shows the relationship between tank topping rate as a function of $\frac{x}{k}$. This relationship covers any type of insulation. It can be seen that, without insulation, the tank topping rate nearly equals the tank fill rate. Insulation is required to prevent the formation of liquid air on the tank outer wall. The minimum insulation thickness is that which prevents the formation of liquid air. Assuming that 1 in. of frost is formed during tank fill, and by trading insulation weight cost against cost of hydrogen lost through boil-off, a x/k ratio of 2.22 (in^2-hr-°F Btu) nets a boil-off rate of 325 gpm.

From a survey, it was determined that foamed polyurethane, with a k of 0.09 Btu/hr-ft^2-°F/in., could be applied in a layer 0.2-in. thick to the exterior of the tank. No vapor barrier is required during application. The density of polyurethane foam is from 2 to 4 lb/cu ft. It must be noted that polyurethane has an impact detonation on sensitivity of 64 ft-lb in the presence of liquid oxygen. This is considered no problem because the foam is bonded to the tank wall and thick enough to prevent the formation of liquid air.

V, E, Propellant Tankage Loading System (cont.)

2. Oxygen Tank

With an assumed fill time of four hours, a use factor of 2, and tank volume of 276,038 gal, the fill rate for the oxygen tank is 2,300 gpm. Prior to fill, the tank is purged with dry nitrogen.

The tank is filled through one 8-in. valve coupling. The design of the components is identical to those described for the fuel tank. After fill, the tank level is maintained by topping at a rate of 50 gpm. No insulation is required for the oxidizer tank.

3. Auxiliary Hydrogen Pressurization Tank

Assuming a fill time of one hour with a use factor of 1.5 and a tank volume of 44,100 gal, the fill rate required for the auxiliary hydrogen tank is 1,100 gpm.

The fill and vent system design, as well as procedural operation, is essentially the same as that for the main hydrogen tank, except for the size of the components. The tank is filled through one 2.5-in. valve, which is also used as the topping fill valve. The main tank vent valve is 6-in. in diameter and is manually disconnected after the filling operation. A 1-in. valve, equipped with a pressure relief feature, is used for venting during the topping cycle.

V, Booster Design Studies (cont.)

F. FACILITIES AND GROUND SUPPORT EQUIPMENT

Only those major differences and conditions affecting the design of facilities for the pressure-fed system as compared with a similar sized pump-fed system, are discussed here.

1. Helium Gas Storage

An extensive high-pressure helium gas storage and pressure-regulation system is needed at the engine test and launch facility to support the pressure-fed rocket system at the 6,700,000-lb thrust level.

Figure V-31 shows a schematic of such a facility. Helium gas would be stored in eighteen 6,000 psi, 338,000 scf gas receivers at 3,000 psi (27,600 lb weight), forming a cascade system. The cascade is actually sized for 3,100,000 scf; this allows for 10% leakage, which is inherent in high-pressure helium storage systems.

Such large quantities of helium can be supplied at the site by tank cars owned by the Bureau of Mines Helium Activity on a government-allocated basis. These railroad cars are charged with 215,000 scf of gas at 2,600 psi. Approximately 15 cars would be required per launch. The storage system would also need five 250 cfm compressors to take the gas at tank car pressure (2,600 psi) and boost it at 6,000 psi to the cascade. From the cascade, a two-step regulation system would supply helium at 3,000 psi to the vehicle helium vessel. Note that approximately 3,100,000 scf of helium is required per launch, while the entire national production is 500,000,000 scf annually.

V, F, Facilities and Ground Support Equipment (cont.)

2. Propellant Use Factor

Use factor is defined as a measure of the heat capacity of a cryogenic system, and is given by the expression:

$$\text{Use Factor} = \frac{\text{Propellant vaporized during fill} + \text{Propellant used for pressurization and combustion}}{\text{Propellant used for pressurization and combustion}}$$

Because of the greater tank wall thickness of pressure-fed systems as compared with pump-fed systems, a larger propellant boil-off, or vaporization of cryogenic propellant, will result during the cool-down and filling phases of operation. This occurs because of the greater heat capacity of the thicker-walled tank mass.

The use factor for liquid oxygen tankage is not pertinent to this discussion because of the availability and relative low cost of liquid oxygen.

Considering the liquid hydrogen tanks only, the use factor for the thin-walled pump-fed tank is 1.15, while for a pressure-fed tank the use factor is 1.33, which indicates that the pressure-fed tank requires 15.3% more liquid hydrogen during the cool-down and filling operation.

3. Research and Development Facilities

The segmental engine concept of arranging 32 thrust chambers around a central external expansion plug is conducive to research testing and development.

V, F, Facilities and Ground Support Equipment (cont.)

Existing facilities may be modified to accommodate the configuration of the segmental thrust chambers, both singularly and in small clusters. Test stands at the Liquid Rocket Plant can test a cluster of eight 210K thrust chambers (the size of one quadrant) firing vertically on to water film-cooled deflector plates.

Adequate liquid-oxygen storage and production facilities are available, but additional liquid-hydrogen storage would be needed for a complete development program. It is assumed that liquid hydrogen will be furnished from government plants.

The high-pressure helium gas facility at the Liquid Rocket Plant would be augmented to supply facilities for testing pressure-fed systems and components.

Existing flow and calibration facilities are adequate for flow-testing with water and liquid oxygen, but additional facilities would be required for hydrogen flow calibration.

Segmental thrust-chamber design allows the use of existing dynamic test facilities for determining vibration modes and frequencies; however, a new facility will be required for determining the slosh and vortexing characteristics of vehicle configuration tankage.

Altitude testing, aerodynamic heating, and model wind tunnel testing will be performed at installations such as the Arnold Engineering Development Center at Tullahouma, Tenn. (AEDC); but equipment (i.e., tankage and piping systems) must be provided by the customer (Aerojet) in order to use these facilities.

V, F, Facilities and Ground Support Equipment (cont.)

4. Fabrication Facilities

Segmented thrust chambers and components can be manufactured and fabricated with modifications to existing facilities at the Liquid Rocket Plant. However, tankage and plug subassemblies must be fabricated in conjunction with outside firms, using their facilities that are modified to accept larger subassemblies.

TABLE V-1

PRESSURIZATION SYSTEMS WEIGHT COMPARISON

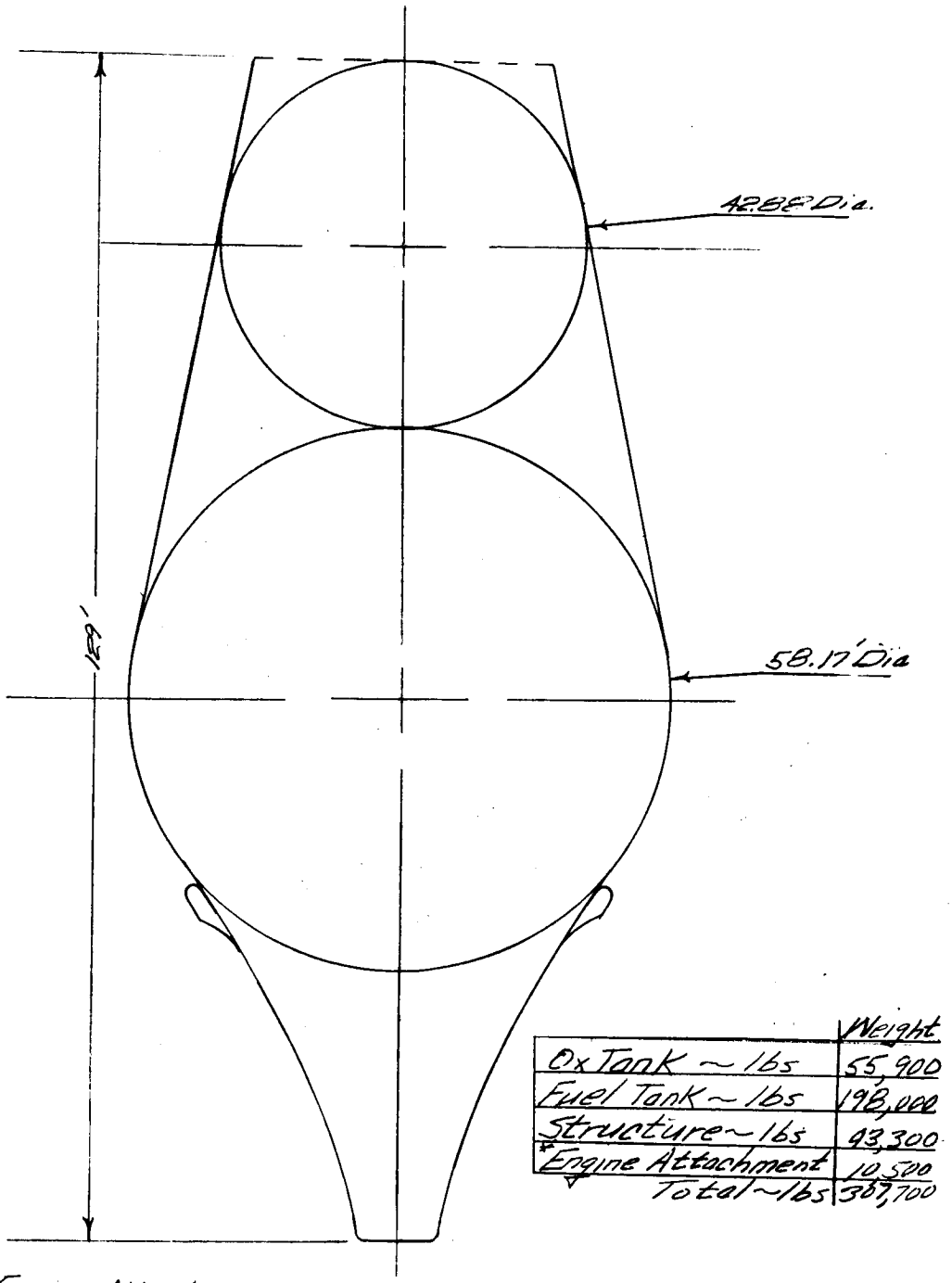
COMPONENT	OXIDIZER SYSTEMS					FUEL SYSTEMS		
	Heated Helium	Heated Oxygen	VaPak	Heated	Bi-Prop Gas Gen	Heated H_2 with Hel Pressur't.	Heated H_2 with Vapor Pressuriz.	Bi-Prop Gas Gen
1. Pressurant Gas								
a. Helium	18,968	----	----	1,906	----	4,376	----	----
b. Hydrogen	----	----	----		713	25,700	37,500	6,470
c. Oxygen	----	137,000	149,000		67,300	----	----	13,050
d. Nitrogen	----	----		109,500	----	----	----	----
2. Pressurant Tanks								
a. Helium	52,000	----	----	5,230	----	12,050	----	----
b. Hydrogen	----	----	----	----	50*	11,000	16,000	5,960*
c. Oxygen	----	7,000*	19,400*	----	7,200	----	----	200*
d. Nitrogen	----	----	----	5,360	----	----	----	----
3. Functional Components and Heat Exchangers	2,960	1,400	100	2,960	200	4,200	500	200
4. Plumbing	410	400	----	400	300	340	200	300
TOTAL SYSTEM WEIGHT	74,338	145,800	168,500	123,356	75,763	57,366	54,200	26,180

* Because pressurant is stored in main propellant tank, weights given are extra tank weight required to contain propellants.

TABLE V-2

CONTROL MOMENT AVAILABLE BY CHAMBER PRESSURE MODULATION

ΔP psi	M^{11} ft kips	M^{1} ft kips	M ft kips	M_{90} ft kips
10	174	37	137	915
20	357	130	227	1,510
30	530	225	305	2,240
40	680	264	416	2,770
50	819	283	536	3,580
75	1,181	293	888	5,920
100	1,540	293	1,247	8,330
125	1,910	293	1,617	10,800

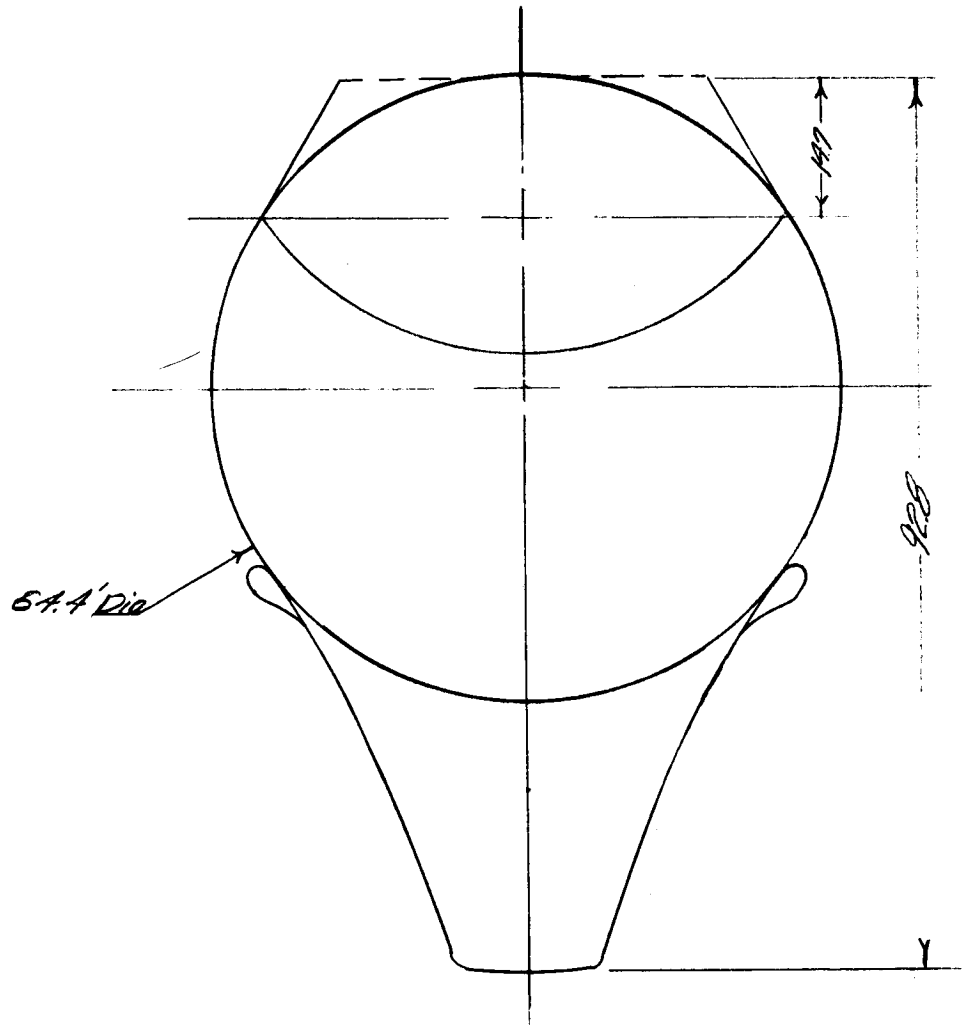


	Weight
Ox Tank ~ lbs	55,900
Fuel Tank ~ lbs	198,000
Structure ~ lbs	93,300
* Engine Attachment	10,500
Total ~ lbs	387,700

* Engine Attachment weight includes required increase in skin-thickness to transmit thrust to thrust structure

Standard Tankage Configuration

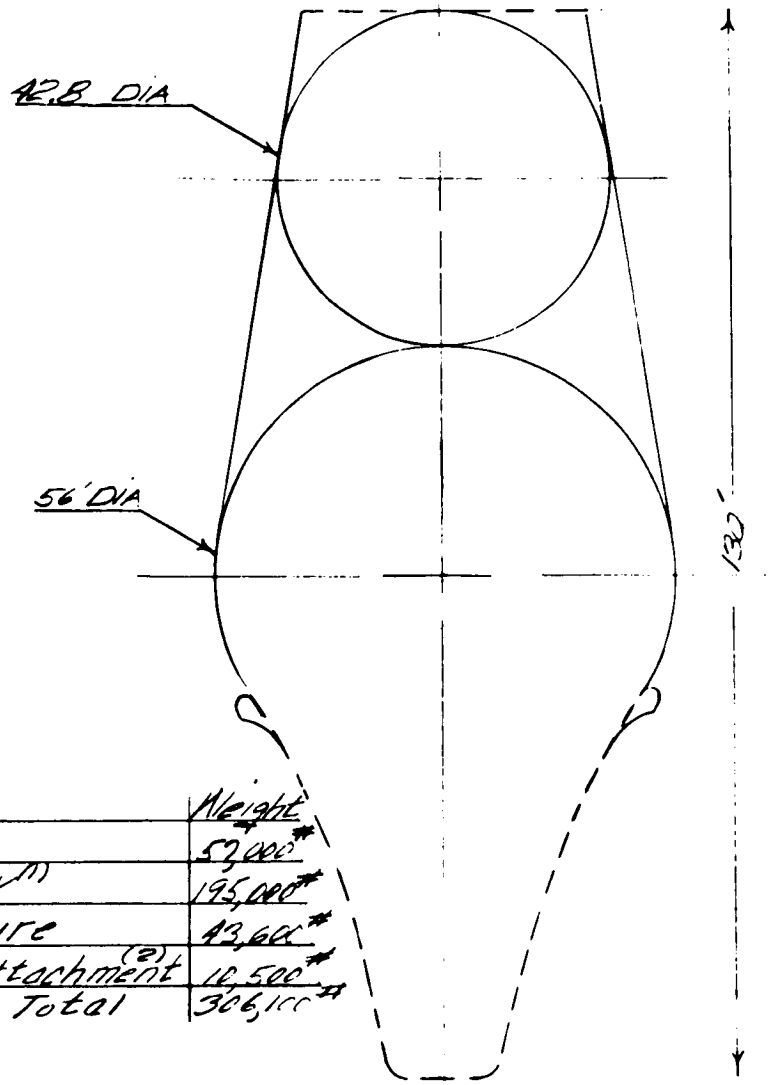
Figure V-1



	Weights
Ox Tank	80,000 #
Fuel Tank	183,000 #
Structure	20,000 #
Engine Attachment	10,500 #
	293,500

Alternative Tankage Configuration--No. 1

Figure V-2



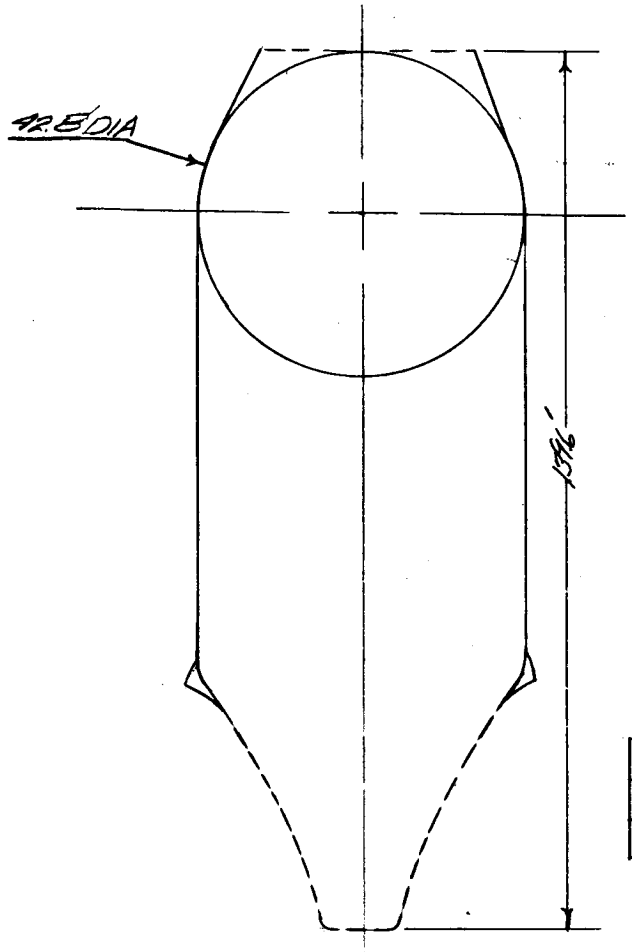
	Weight
Ox Tank	52,000#
Fuel Tank (1)	195,000#
Structure	43,600#
Engine Attachment (2)	16,500#
Total	306,100#

(1) Includes an estimated 70,000# to convert plug volume into pressure container.

(2) Engine attachment weight includes required increase in skin thickness to transmit thrust to intertank structure.

Alternative Tankage Configuration--No. 2

Figure V-3

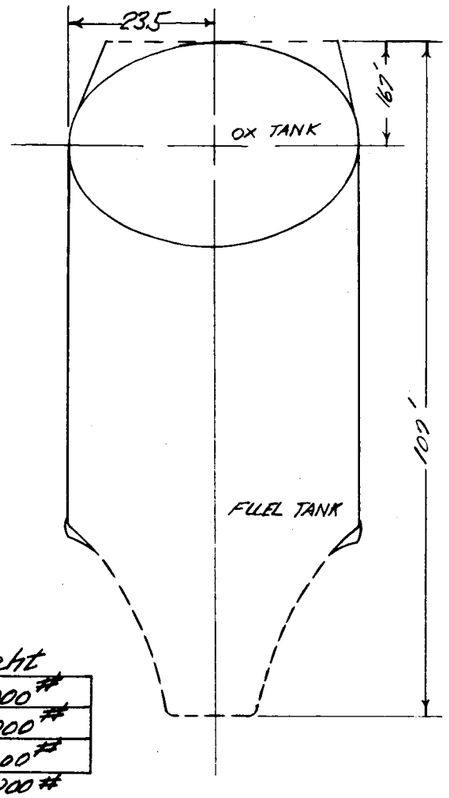


Ox Tank	57,000#
Fuel Tank*	290,000#
Structure	11,000#
Total	358,000#

* Includes an estimated 70,000# to convert plug volume into pressure container.

Alternative Tankage Configuration--No. 3

Figure V-4

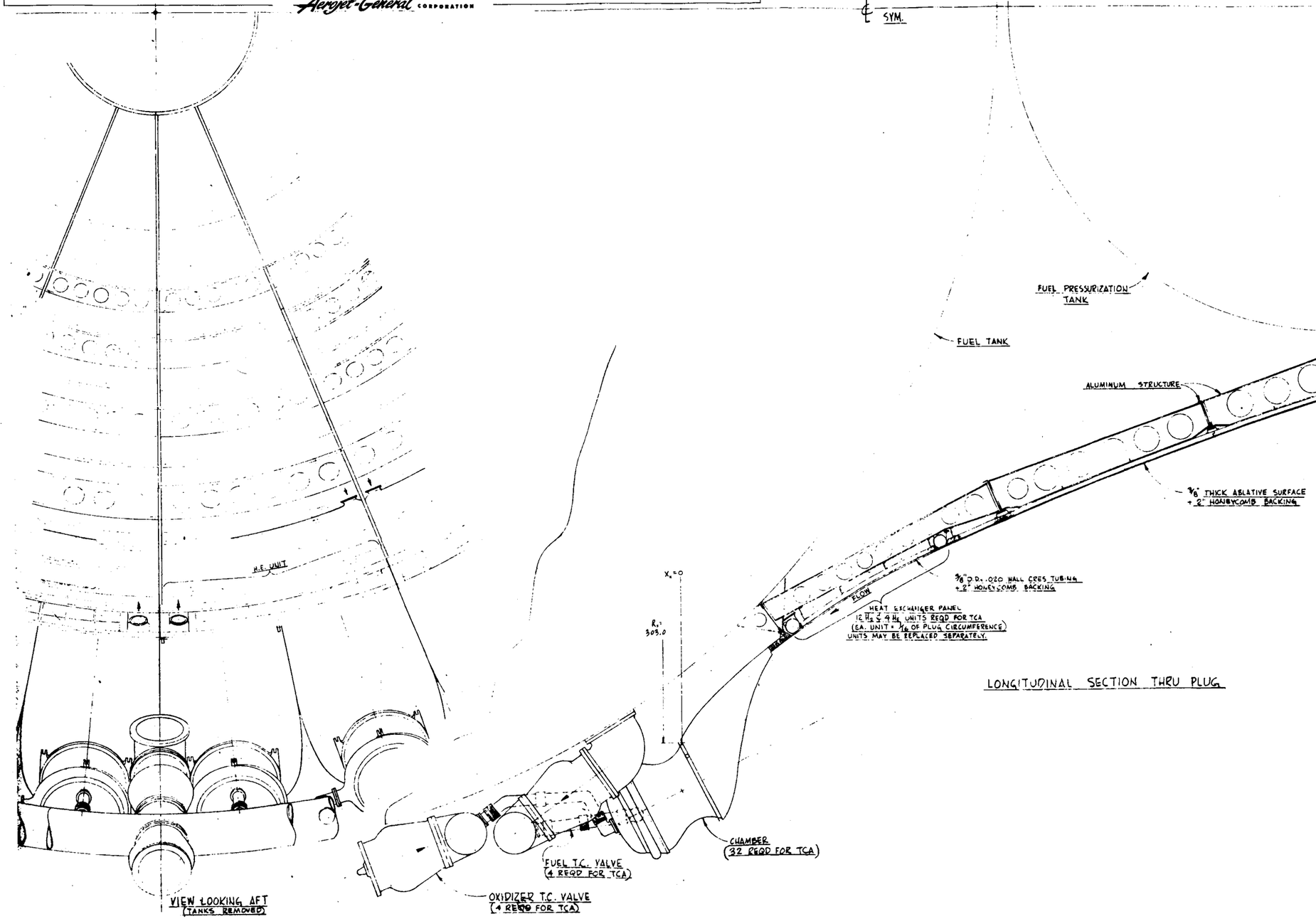


	Weight
Ox Tank	65,000#
Fuel Tank*	320,000#
Structure	11,000#
Total	396,000#

*Includes an estimated 20,000# to convert plug volume into pressure container

Alternative Tankage Configuration--No. 4

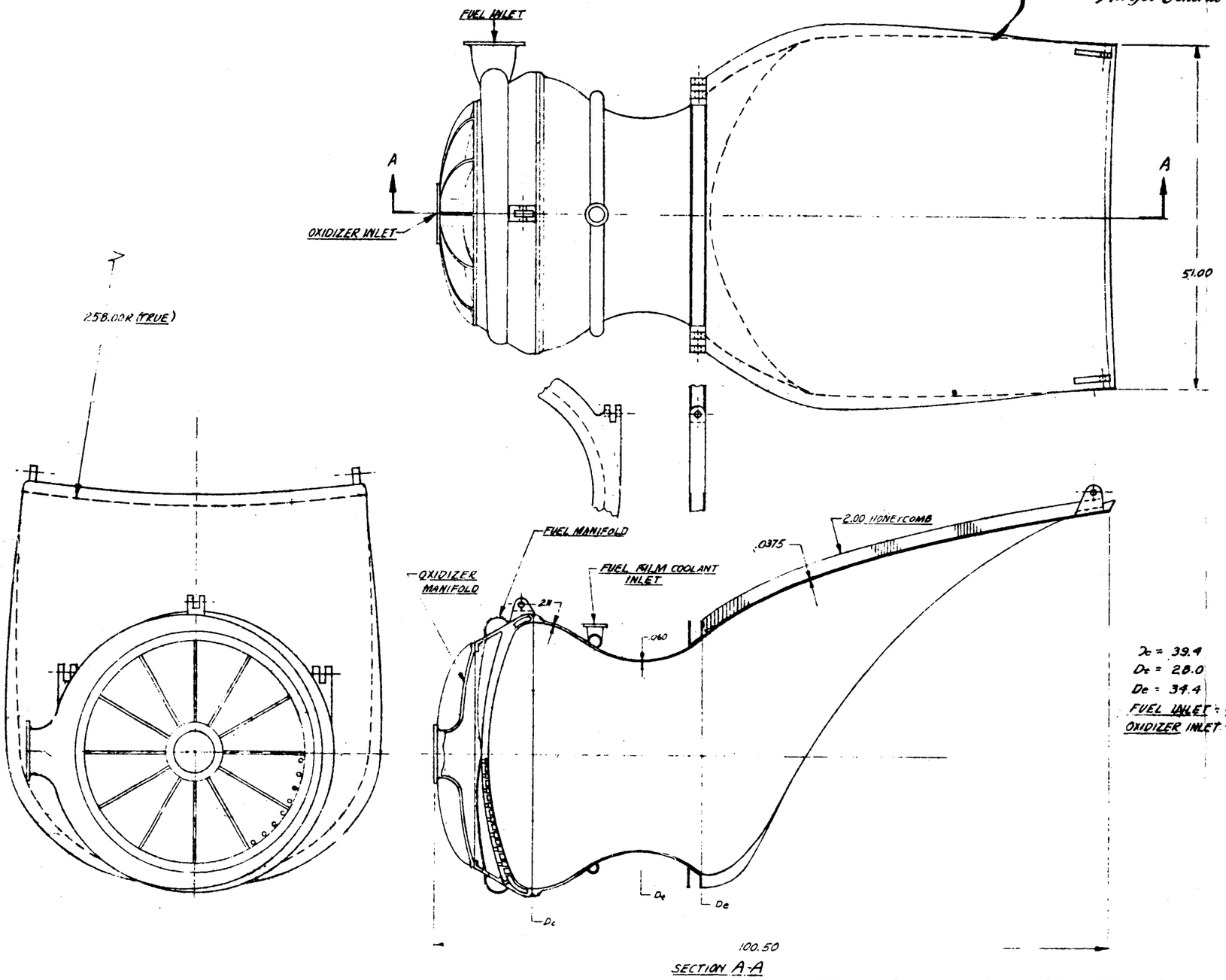
Figure V-5



PLUG CONTOUR	
R_x	X_x
310.0	− 6.80
303.0	0.00
295.0	4.50
286.0	12.30
274.0	25.70
256.0	48.00
234.0	84.50
205.0	147.00
168.0	247.00
121.0	417.00
61.0	732.00
0.0	1340.00

6.7 x 10^6 -lb Plug-Nozzle Thrust Chamber Assembly

Figure V-6



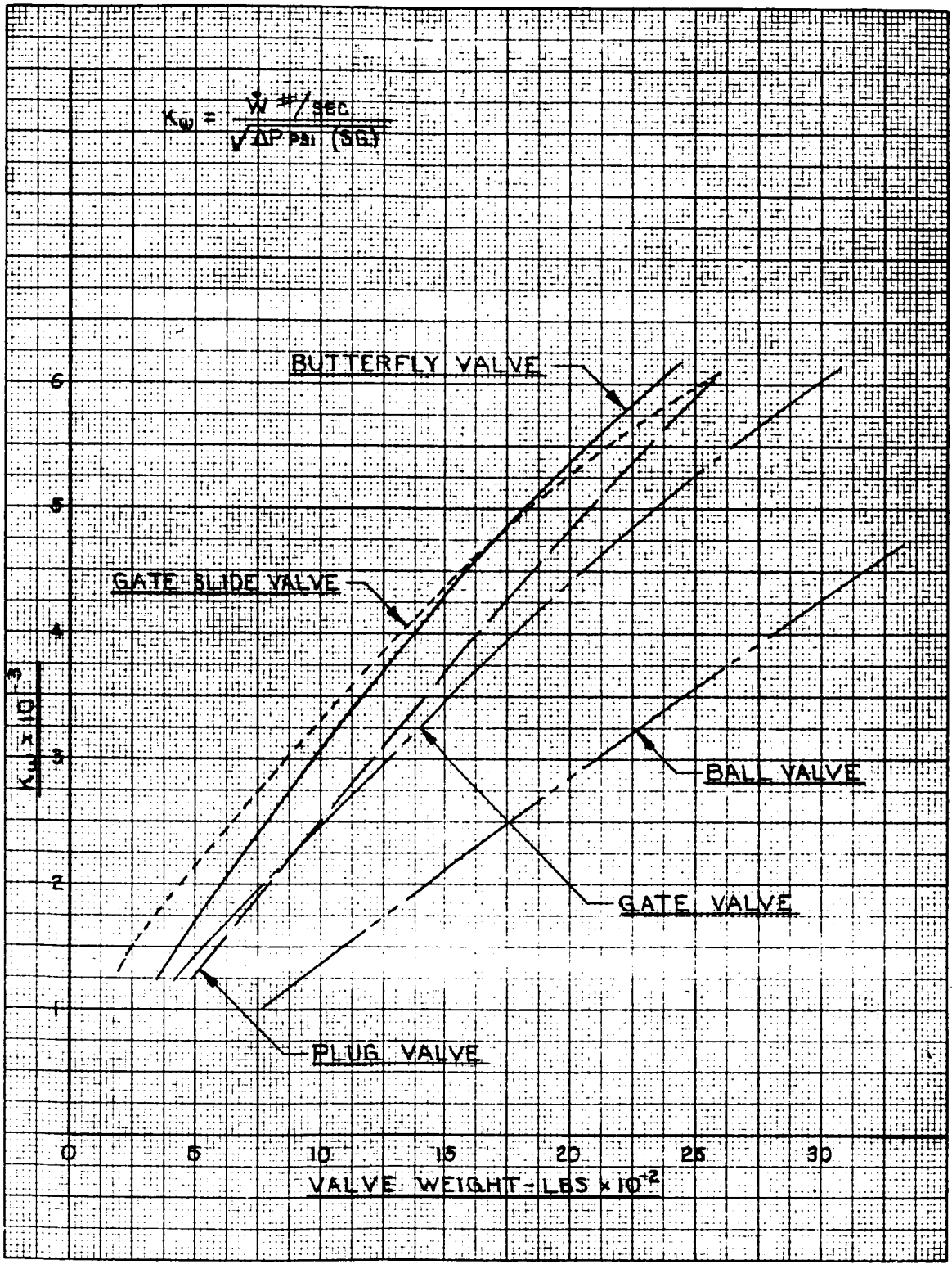
Thrust Chamber Assembly--Film-Cooled, Pressure-Fed Plug-Nozzle Engine

Figure V-7

NOTE: PROPELLANTS LO_2/LH_2 λ= 97% η_c= 95% ISPeff = 363.6 Secs	UNITS		Sea Level	Vacuum		
COMBUSTION CHAMBER						
THRUST	LBS	F	6.7×10^6	8.7×10^6		
PLENUM PRESSURE	PSIA	Pc	250	250		
PRESS. RATIO-INJ. FACE / PLENUM	—	—	1.052	1.052		
INJECTOR FACE PRESSURE	PSIA	Pc'	263	263		
NOZZLE EXIT PRESSURE	PSIA	Pe	14.7	14.7		
NOZZLE THROAT AREA	IN^2	At	19600	19600		
NOZZLE THROAT DIAMETER	IN	Dt	28.0	28.0		
NOZZLE PRESSURE RATIO	—	Pc/Pe	17	17		
NOZZLE AREA RATIO	—	Ae/At	18	18		
NOZZLE EXIT DIAMETER	IN	De	668	668		
THRUST COEFFICIENT	—	Cf	1.30	1.685		
CHARACTERISTIC VELOCITY	FT/SEC	C'*	7500	7500		
EFFECTIVE EXHAUST VELOCITY	FT/SEC	C	9750	12830		
TOTAL PROPELLANT FLOW RATE	LBS/SEC	Wt	22200	22200		
MIXTURE RATIO	—	M.R.	6.0	6.0		
FUEL FLOW RATE	LBS/SEC	Wf	3170	3170		
OXID. FLOW RATE	LBS/SEC	Wo	19030	19030		
SPECIFIC IMPULSE	SEC	Isp	302	390		
CHARACTERISTIC LENGTH	IN	L^*	25	25		
CHAMBER TO THROAT AREA	—	Ac/At	2	2		
CHAMBER DIAMETER	IN	Dc	39.4	39.4		
NOZZLE EXIT CONFIGURATION	—	—	Truncated Plug L = 393 In.			
GAS GENERATOR						
TOTAL FLOW RATE	LBS/SEC	$Wt_{G.G.}$				
MIXTURE RATIO	—	M.R.				
ENGINE						
THRUST	LBS	F				
TOTAL PROPELLANT FLOW RATE	LBS/SEC	Wt				
MIXTURE RATIO	—	M.R				
SPECIFIC IMPULSE	SEC	Isp_{eng}				
TURBINE EXHAUST THRUST	LBS					

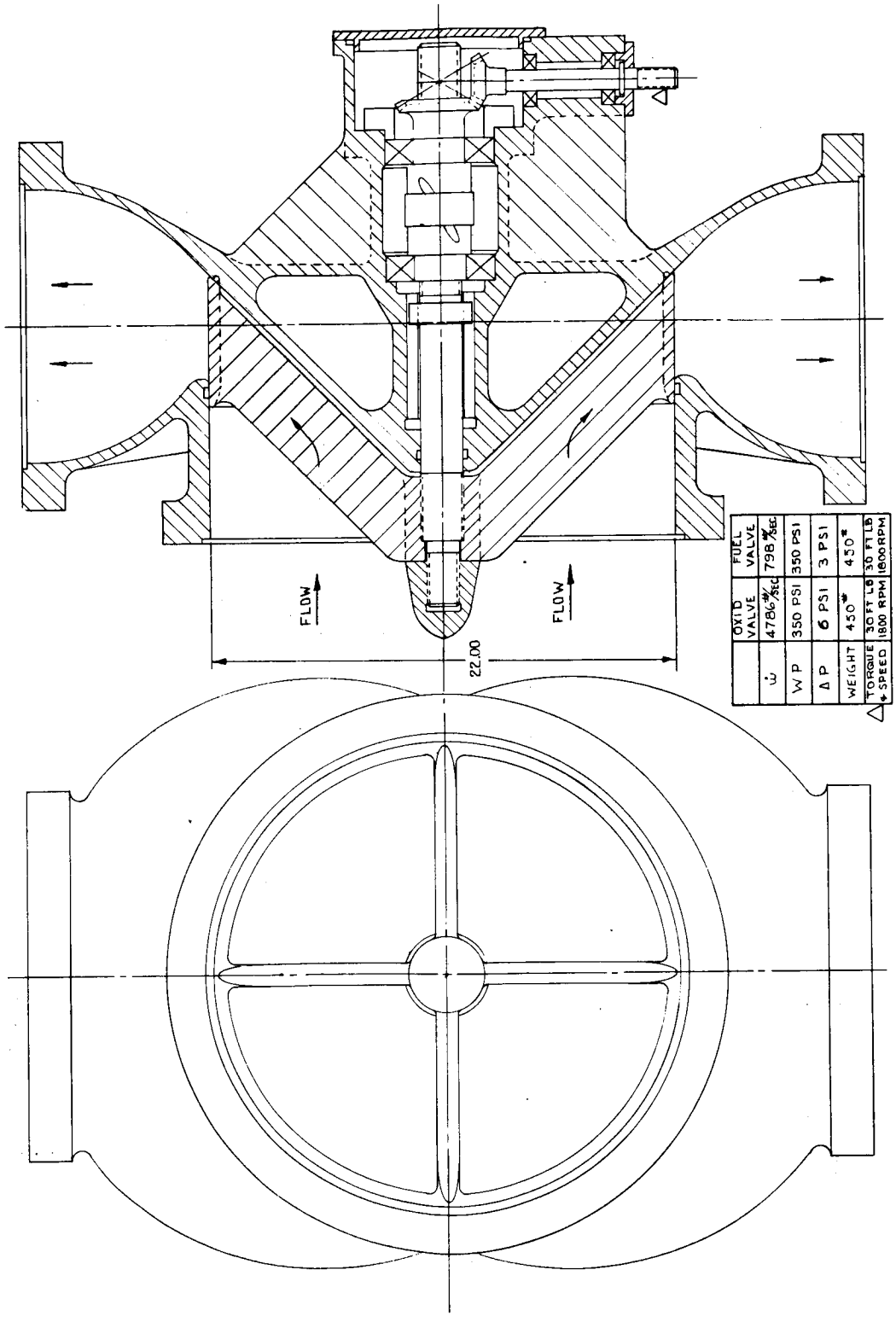
Plug Nozzle--6.7 x 10^6-lb Thrust, 32 Discrete Throats

Figure V-8



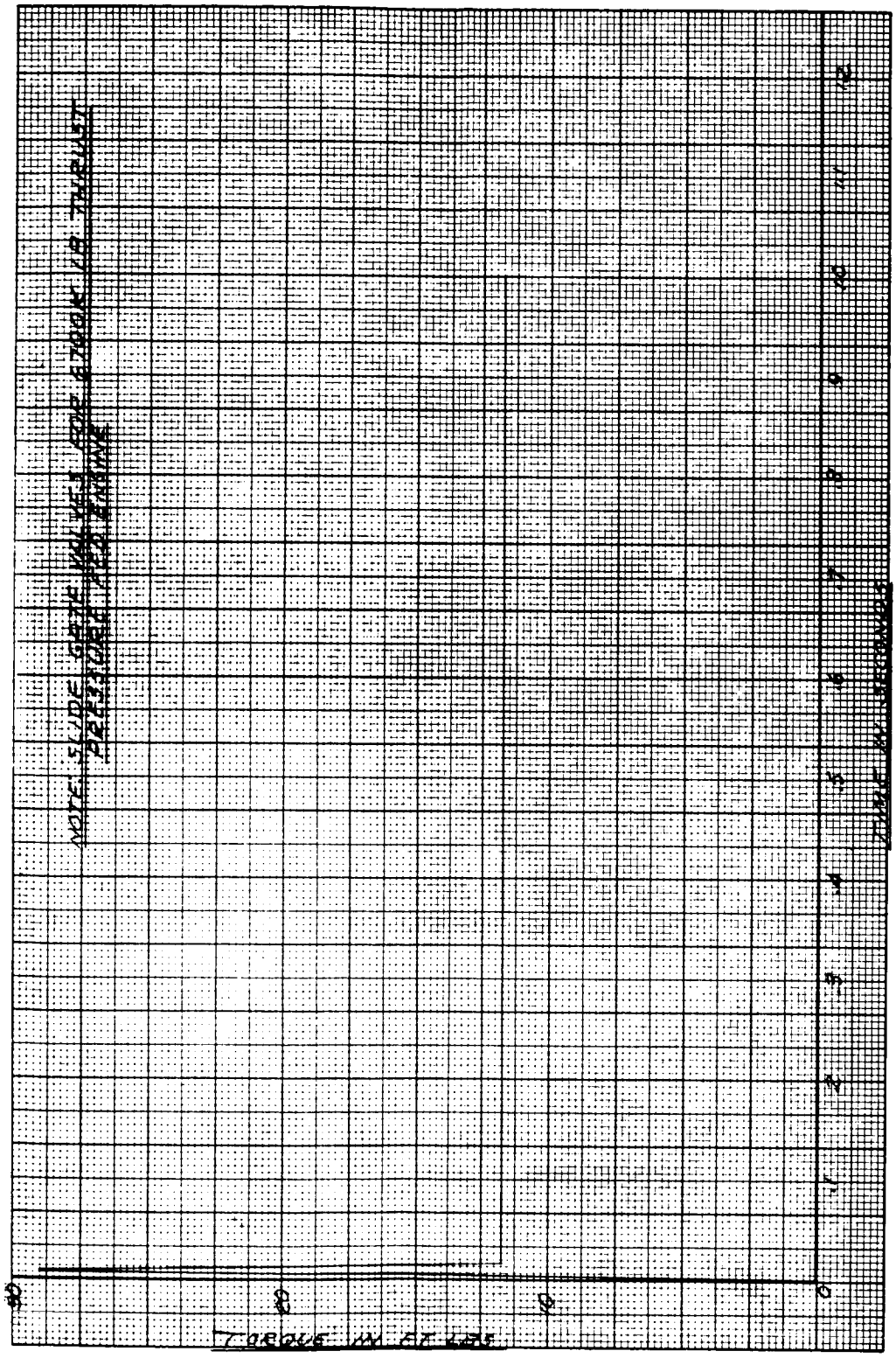
K_W Versus Valve Weight

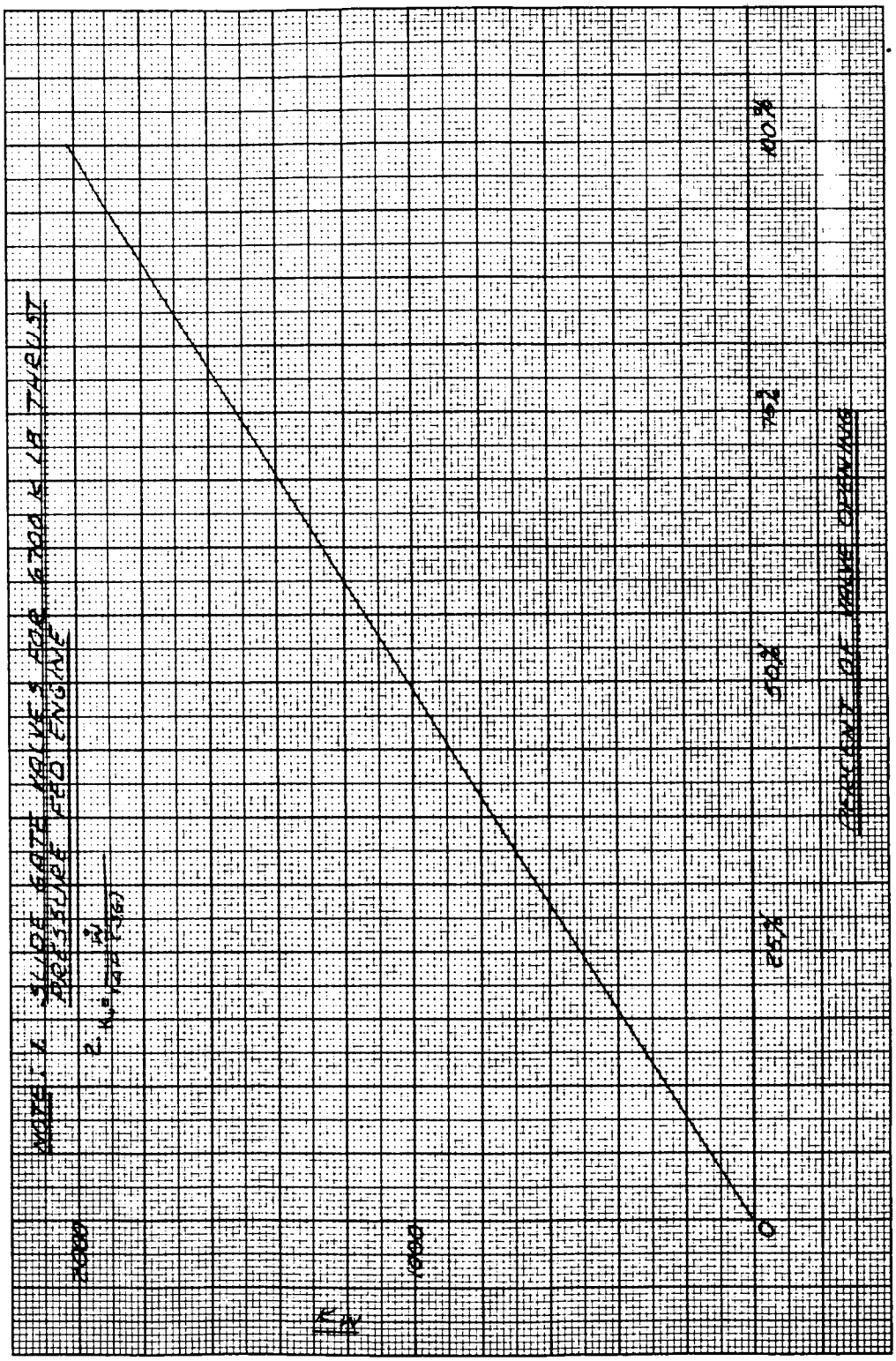
Figure V-9



Thrust Chamber Valves for 6, 700K-lb-Thrust Pressure-Fed Engine

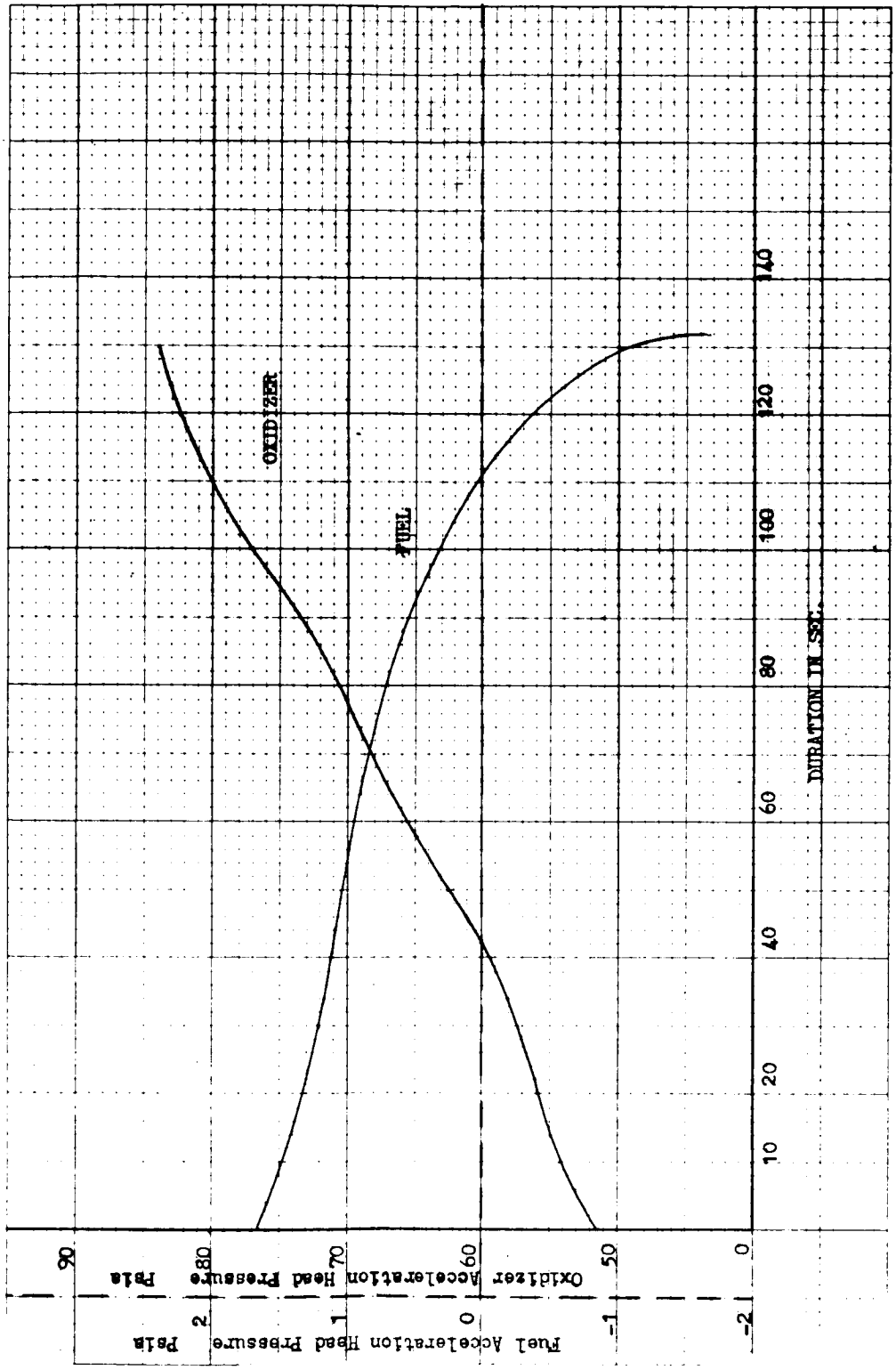
Figure V-10


NOTE: SLIDE GATE VALVES FOR 4,000 LB THRUST
PROPELLANT ACTUATED ENGINE
TORQUE IN FT-LBS
0
20
40
60
80
100
120
140
160
180
200
220
240
260
280
300
320
340
360
380
400
420
440
460
480
500
520
540
560
580
600
620
640
660
680
700
720
740
760
780



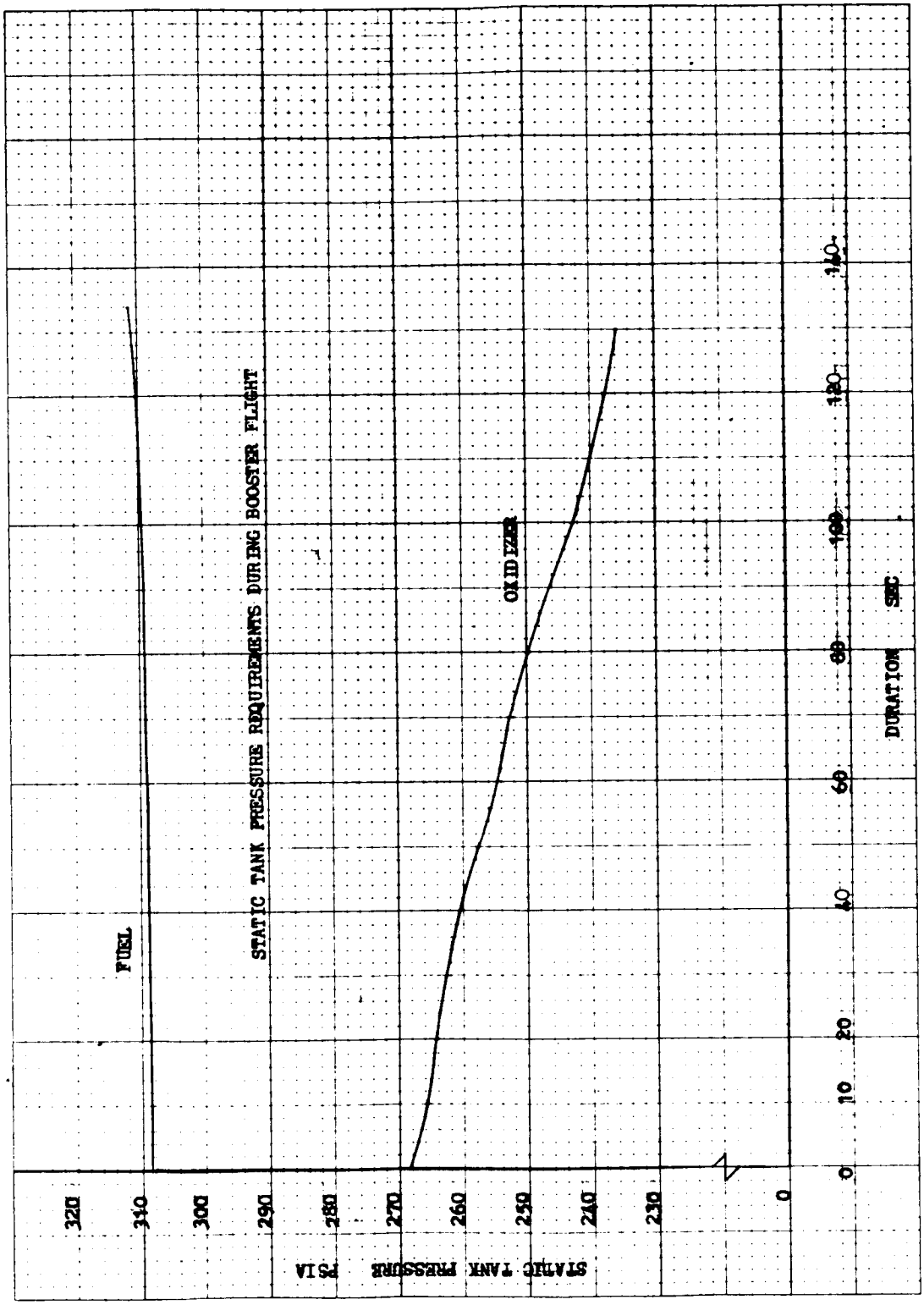
Thrust Chamber Valves K_w Versus Percent of Valve Opening

Figure V-12



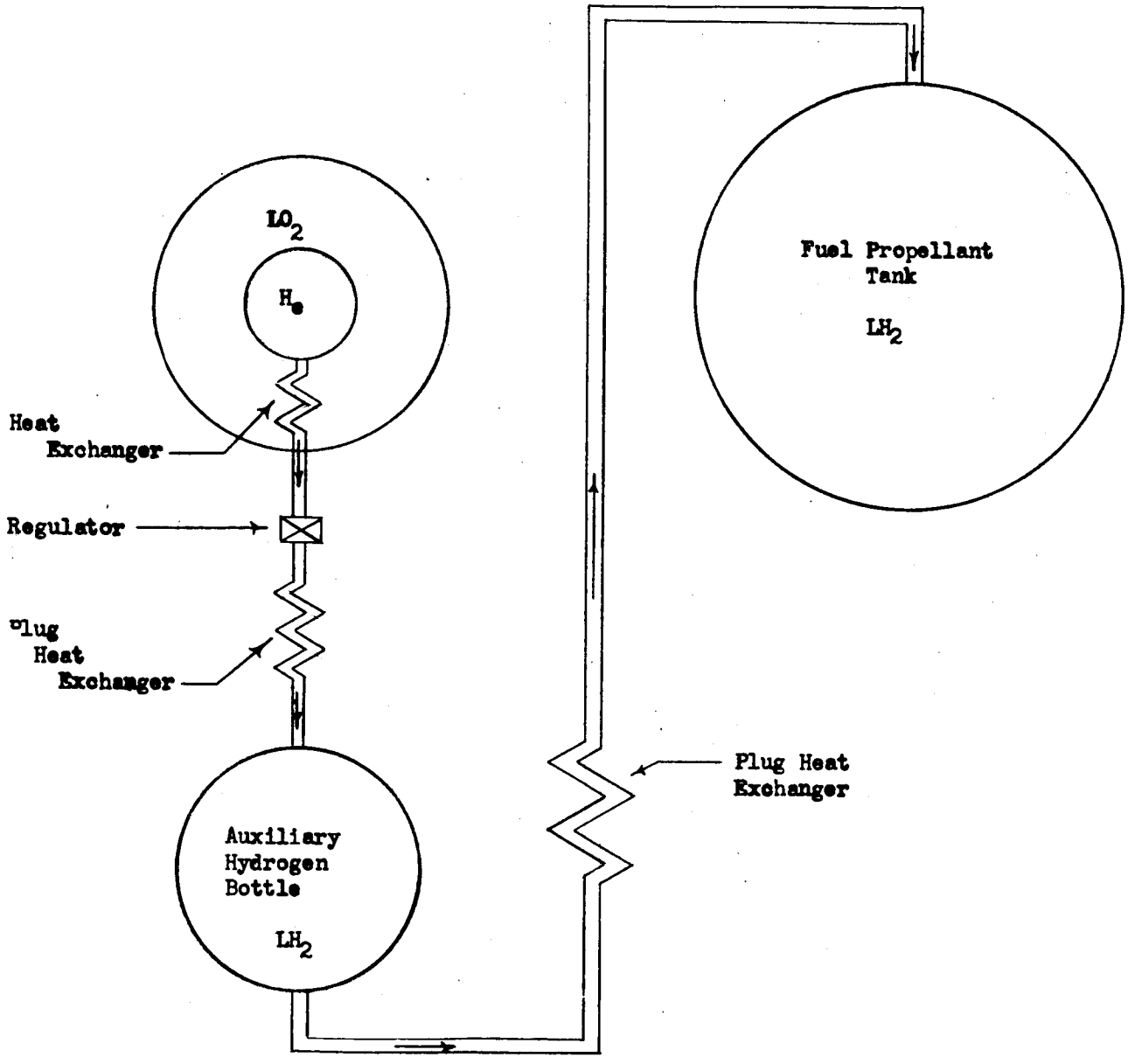
Acceleration Head During Booster Flight

Figure V-13



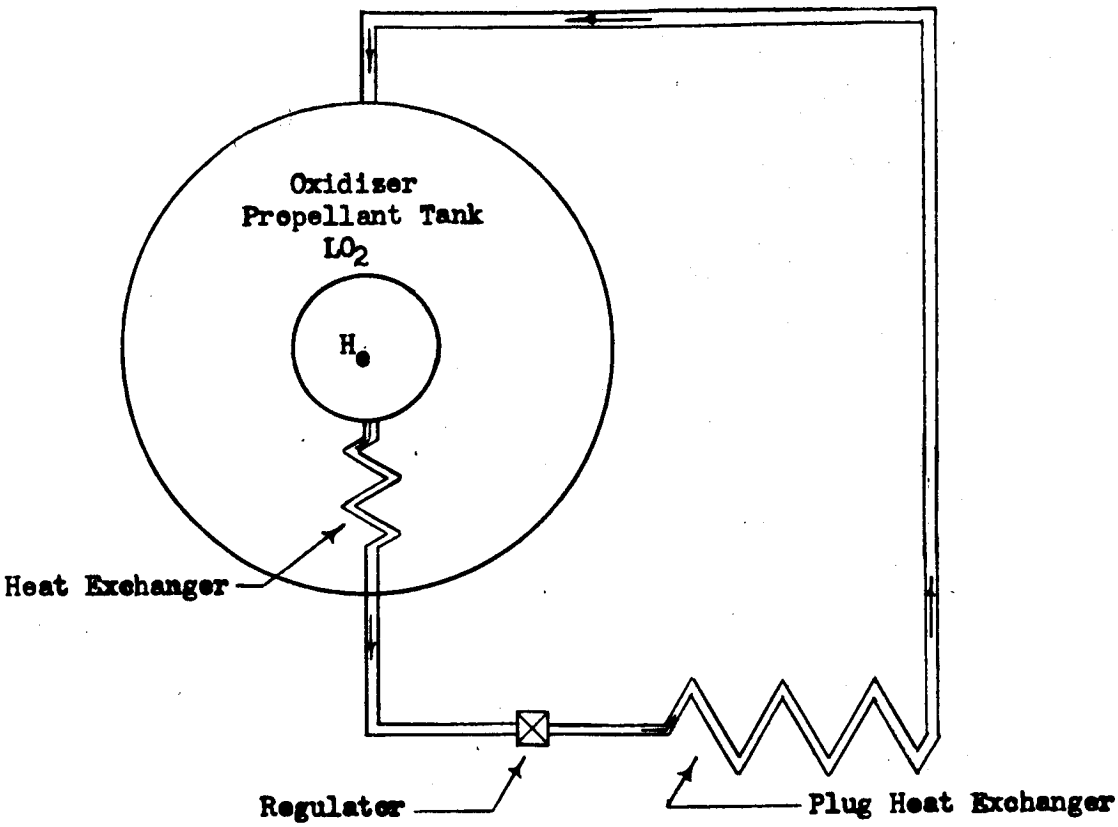
Static Tank Pressure Requirements During Booster Flight

Figure V-14



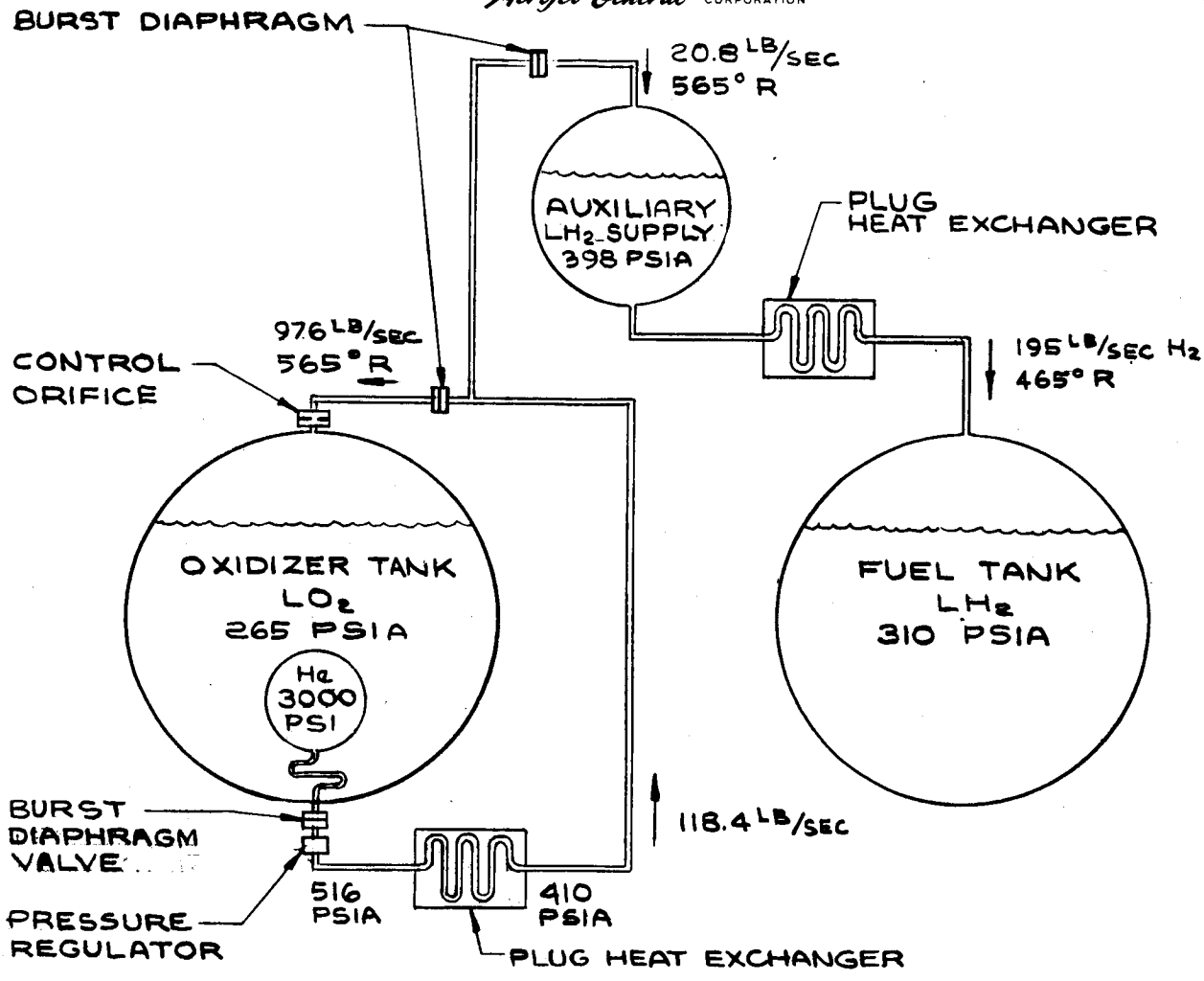
Fuel Tank Pressurization--Heated Hydrogen Pressurized by Heated Helium

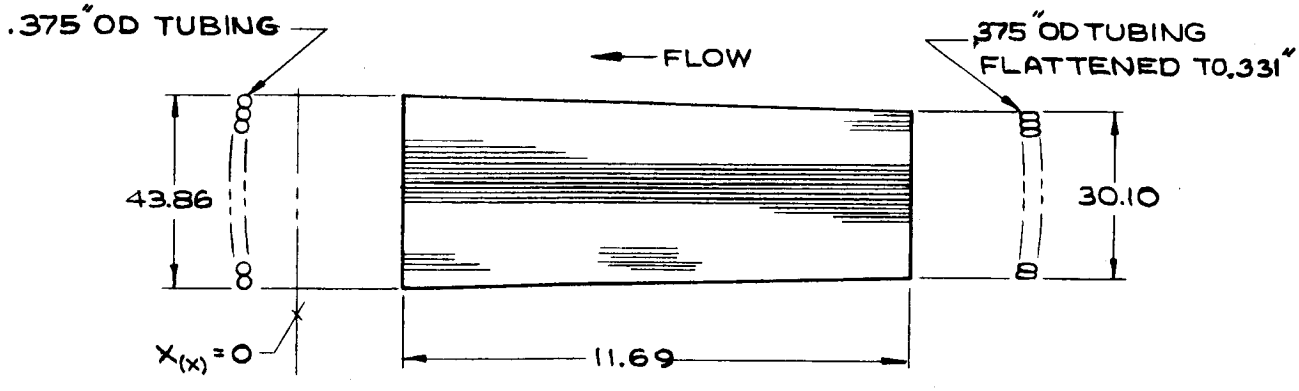
Figure V-15



Oxidizer Tank Pressurization--Heated Helium

Figure V-16

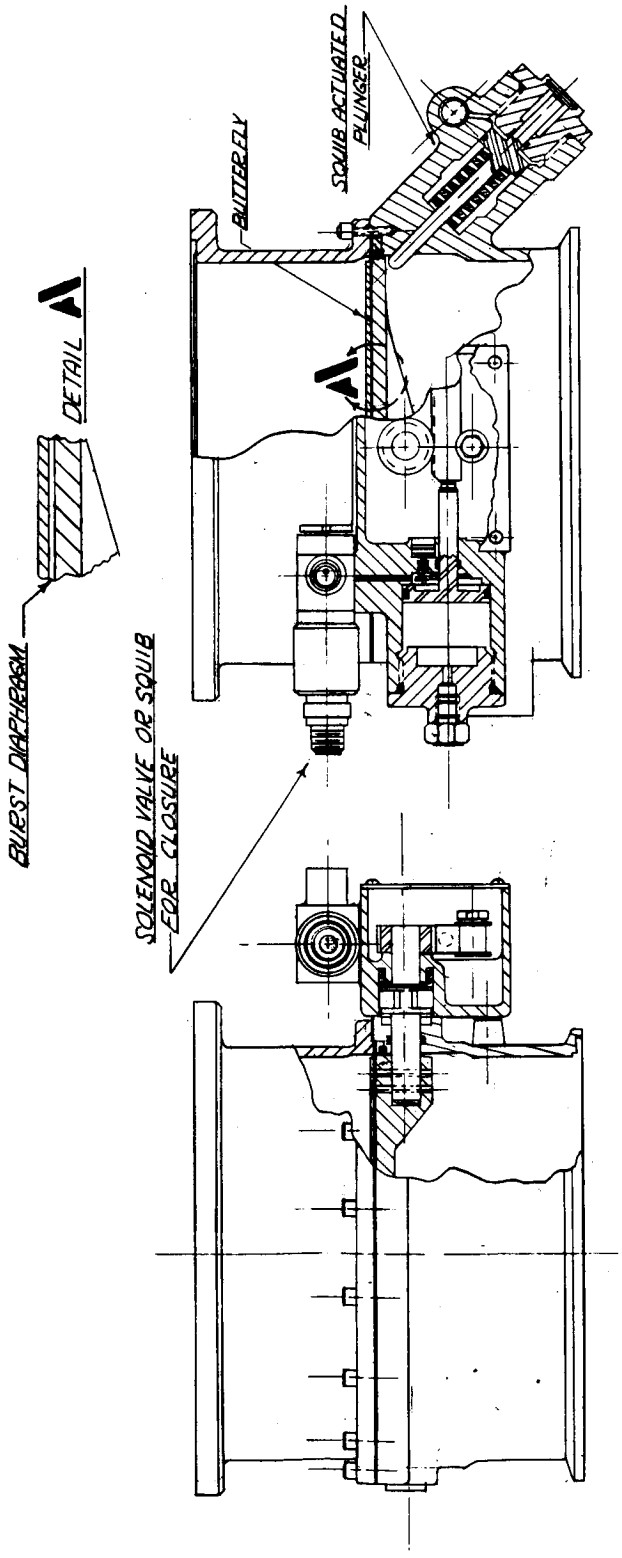




PLUG NOZZLE
HEAT EXCHANGER
SEGMENT

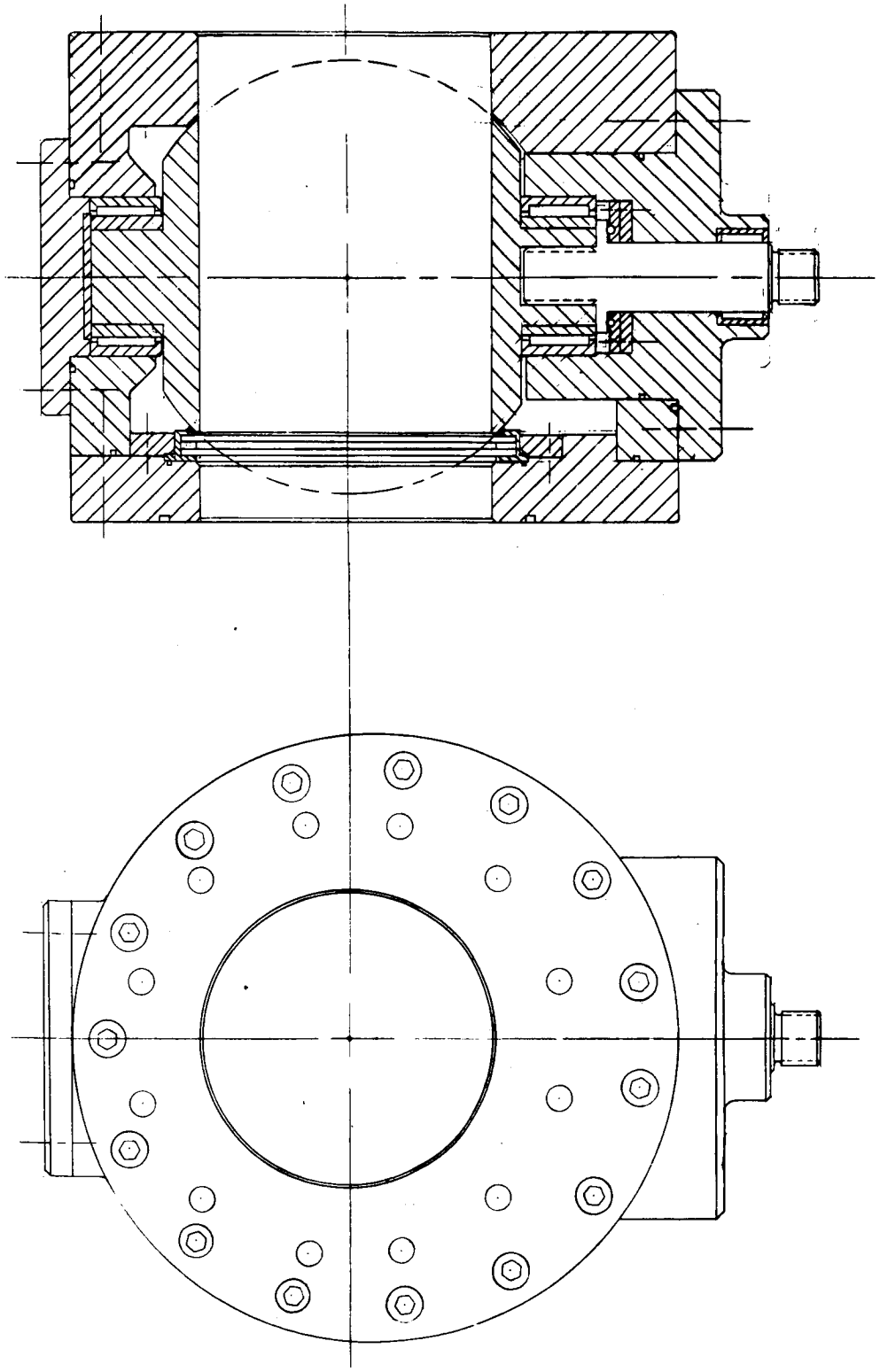
Integrated Pressurization System

Figure V-17



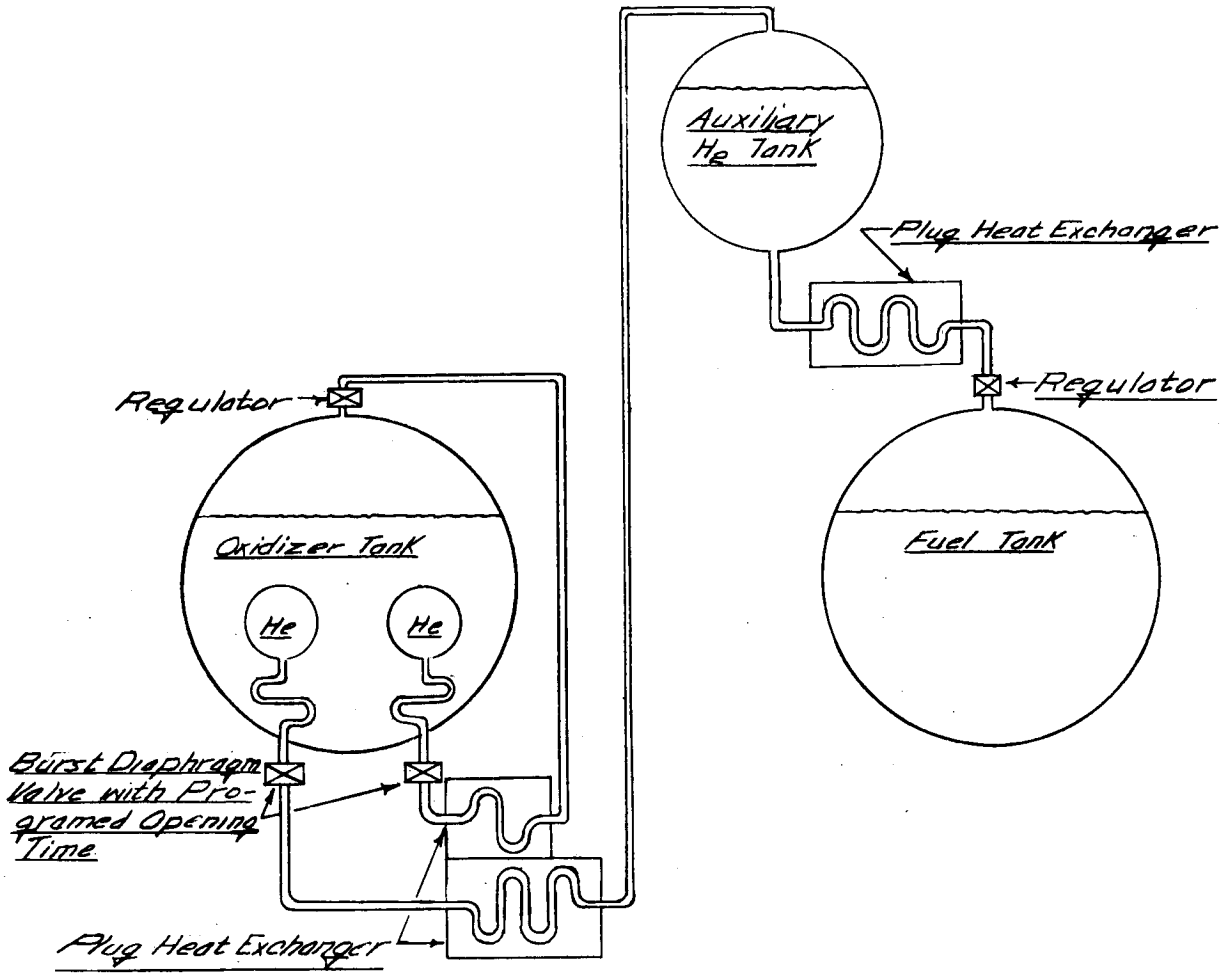
Burst-Diaphragm Valve

Figure V-18



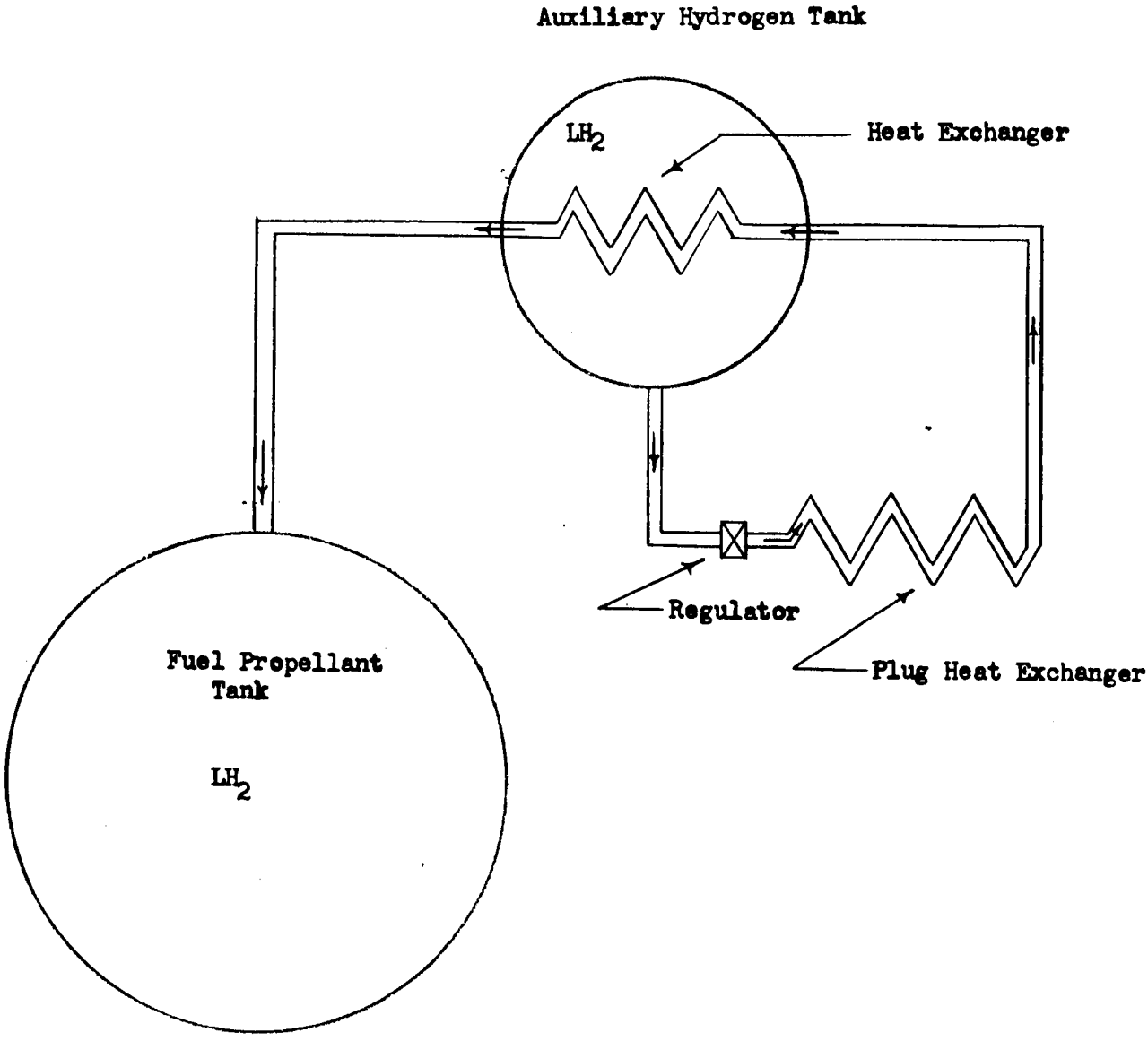
Helium-Pressure Control Valve

Figure V-19



Alternative Pressurization System

Figure V-20



Fuel-Tank Pressurization--Heated Hydrogen Pressurized by Heated Vaporized Hydrogen

Figure V-21

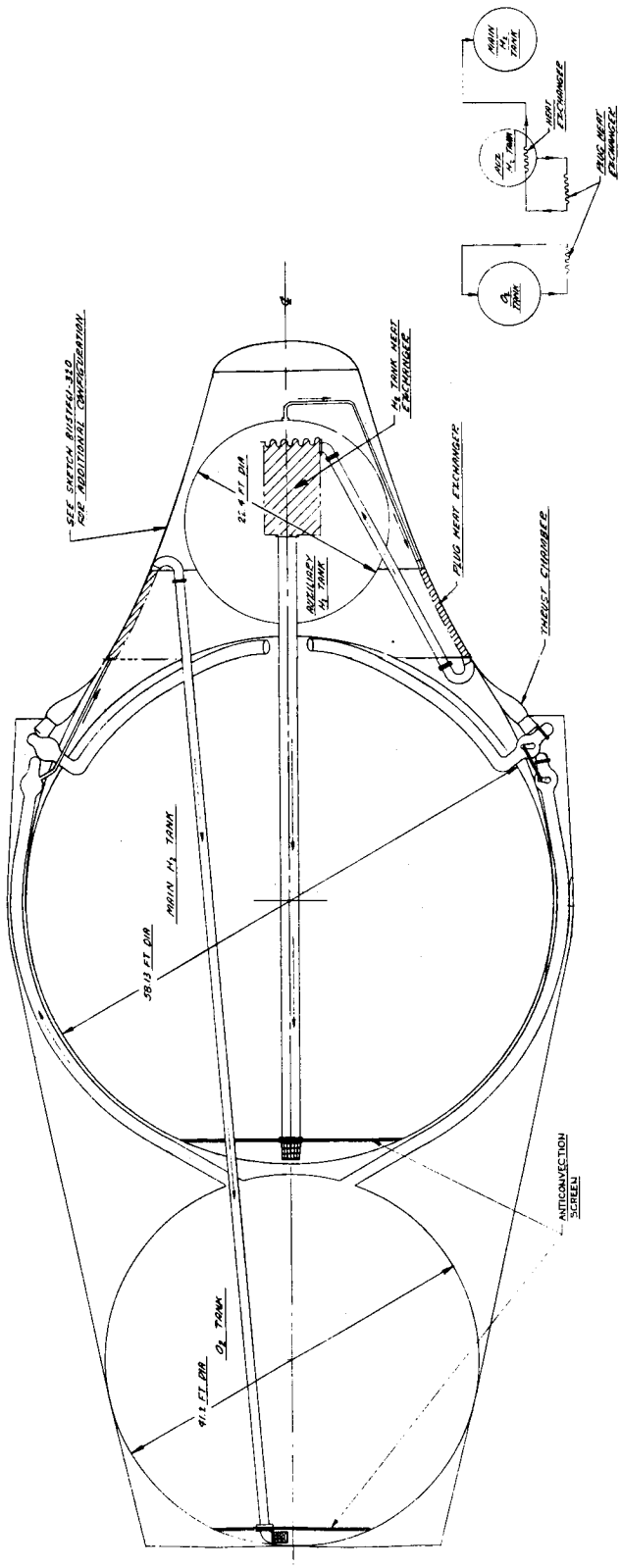
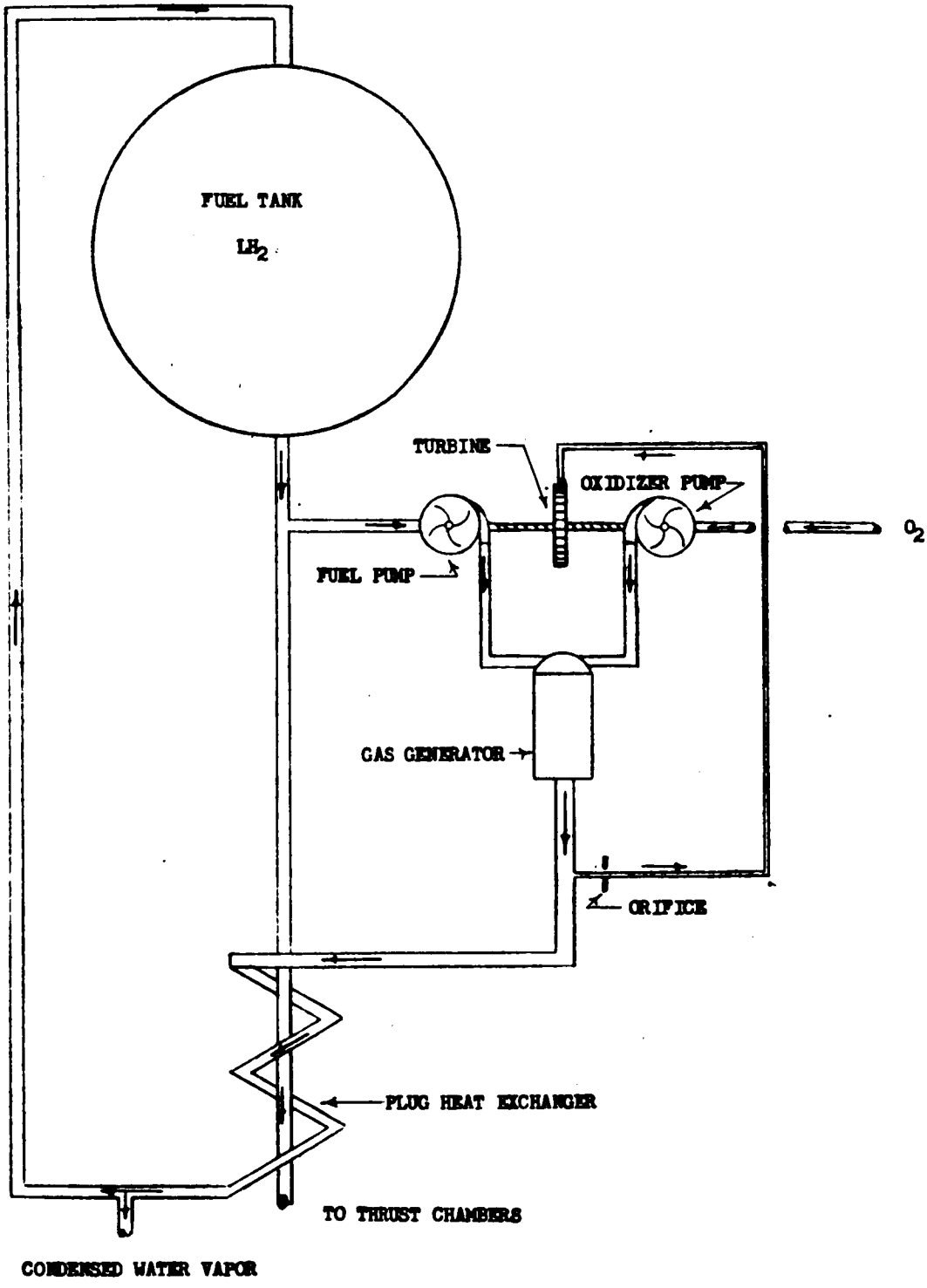


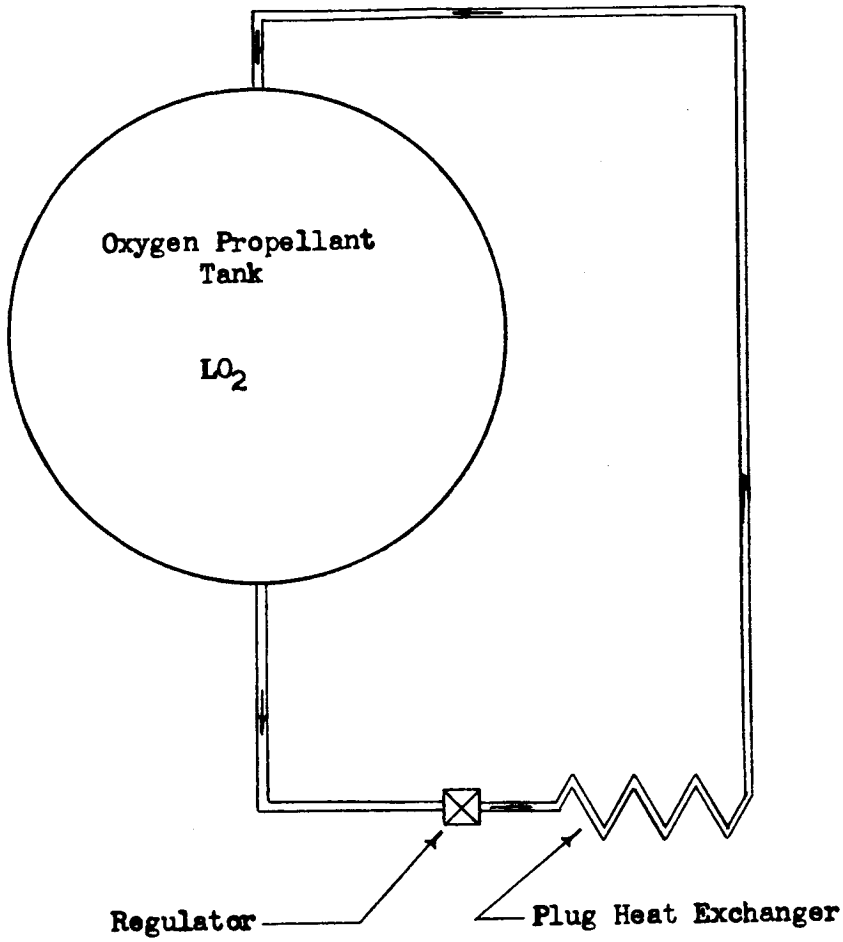
Figure V-22

Booster System with Vaporization of Main Propellants for Pressurization



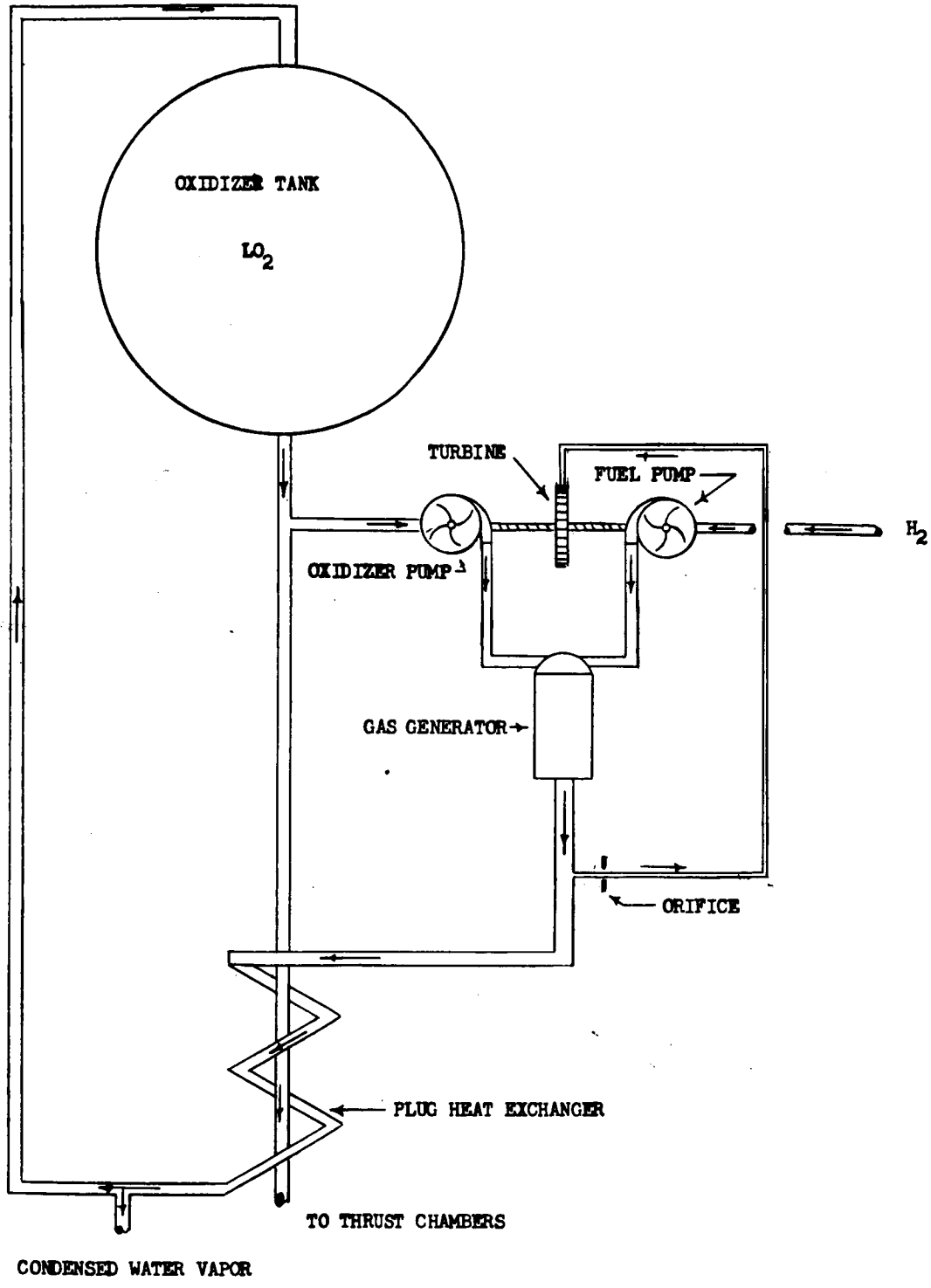
Fuel Tank Pressurization--Exhaust Gases from Liquid Bipropellant Gas Generator

Figure V-23



Oxidizer Tank Pressurization--Heated Oxygen

Figure V-24



Oxidizer Tank Pressurization--Exhaust Gases from Liquid Bipropellant Gas Generator

Figure V-25

VāPak Pressurization System--6 x 10^6-lb Thrust

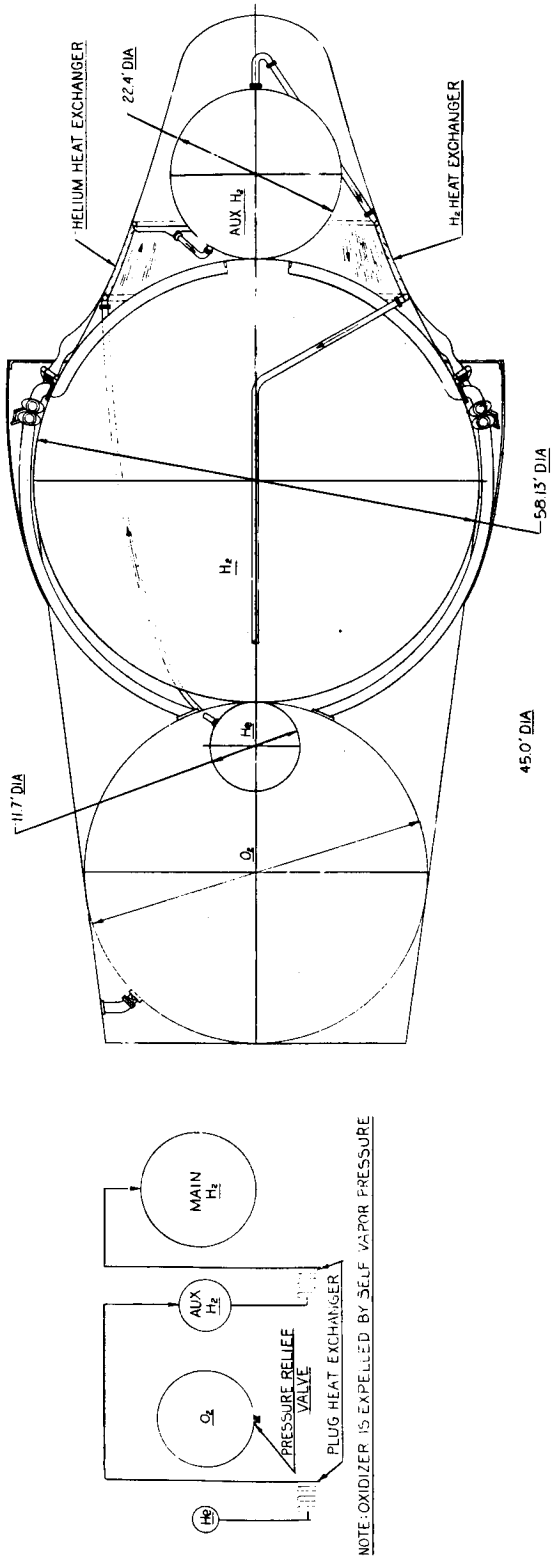
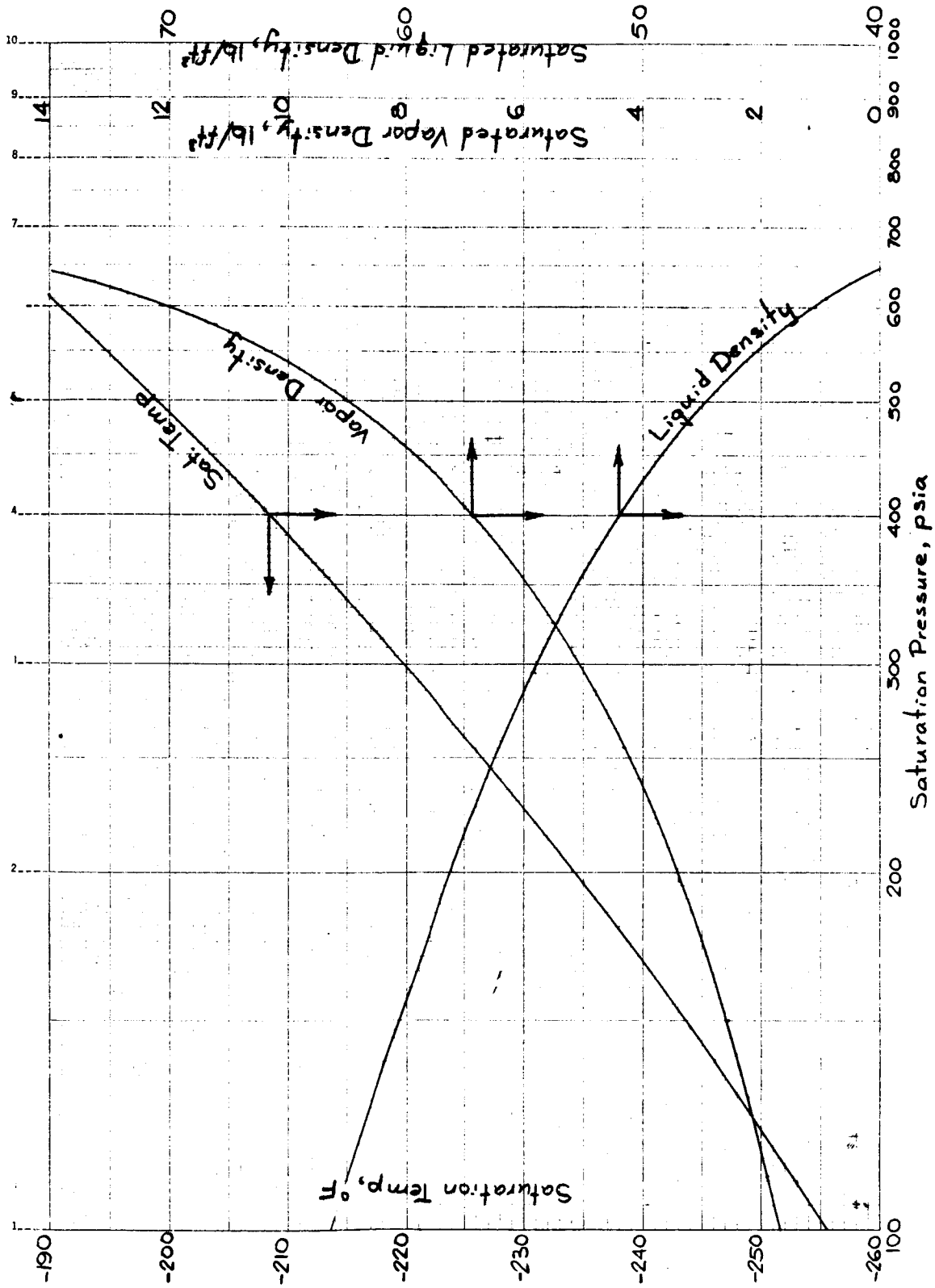
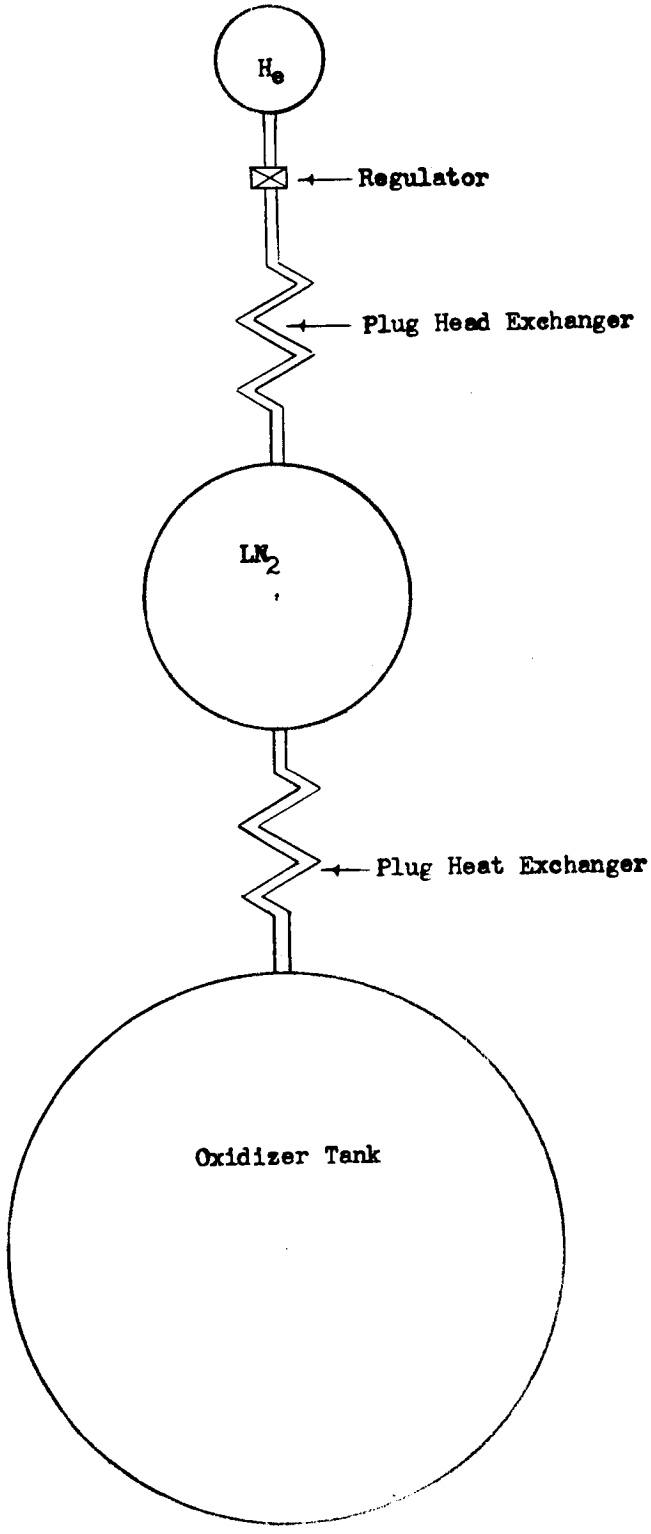


Figure V-26



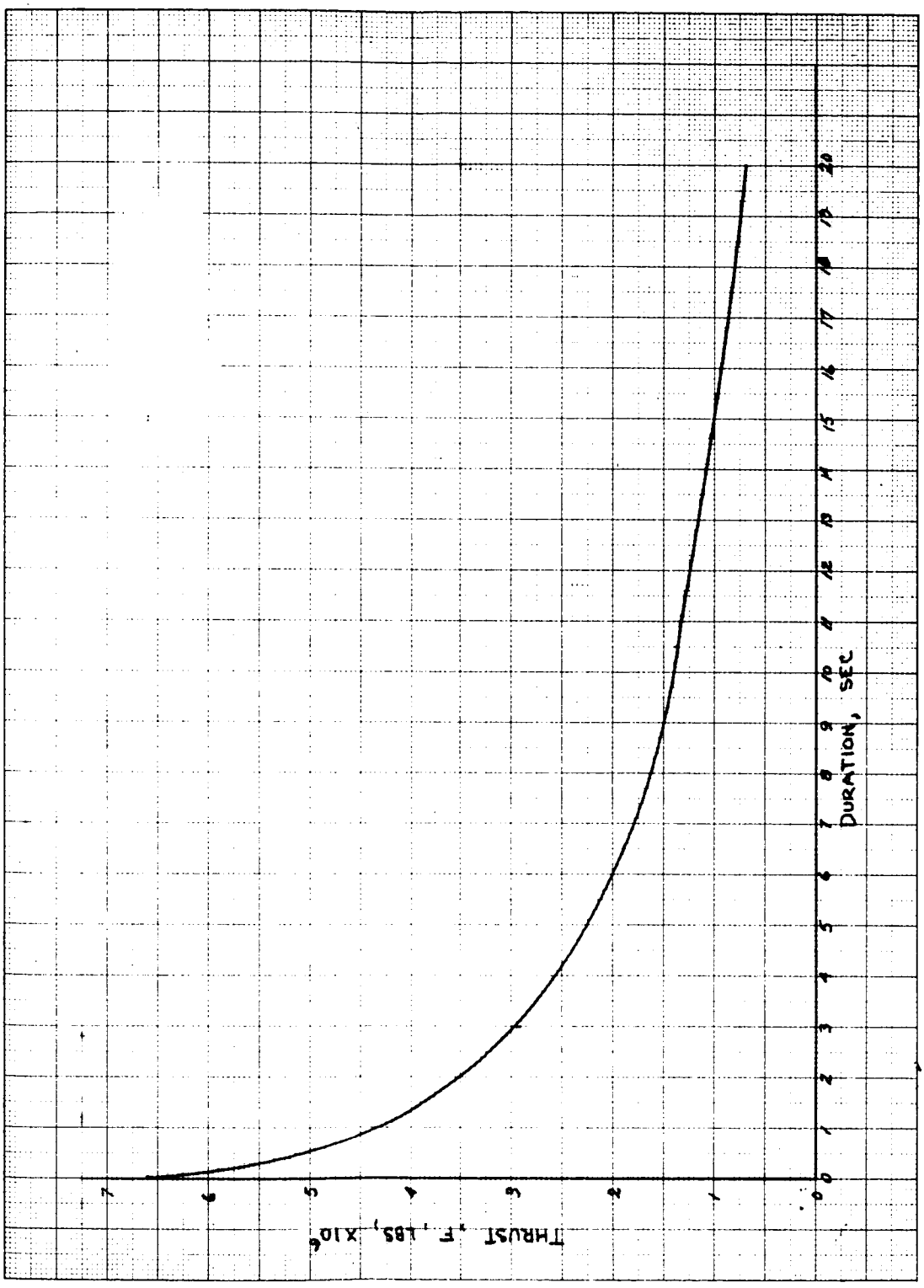
Properties of Saturated O_2

Figure V-27



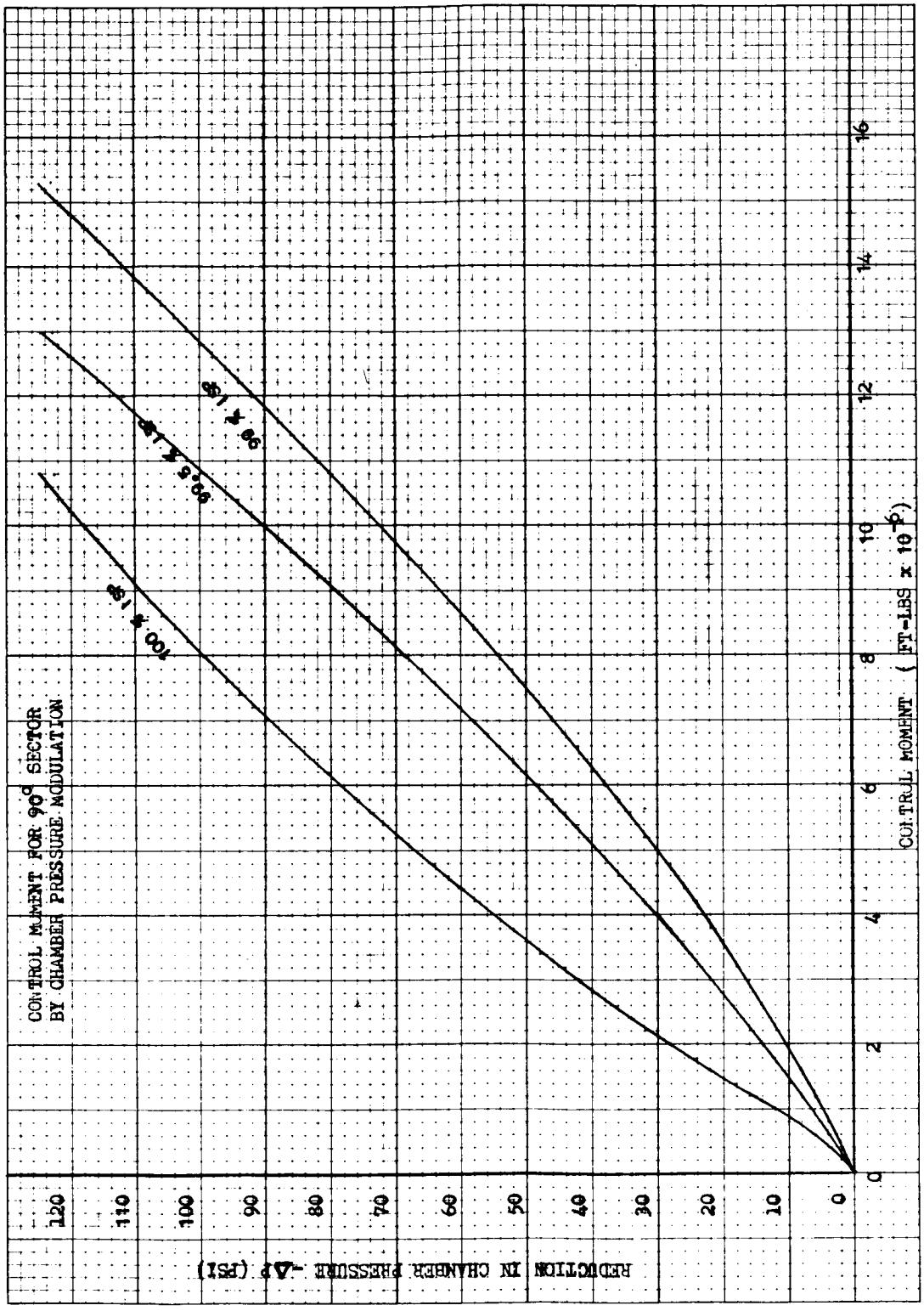
Oxidizer Tank Pressurization--Heated Nitrogen

Figure V-28



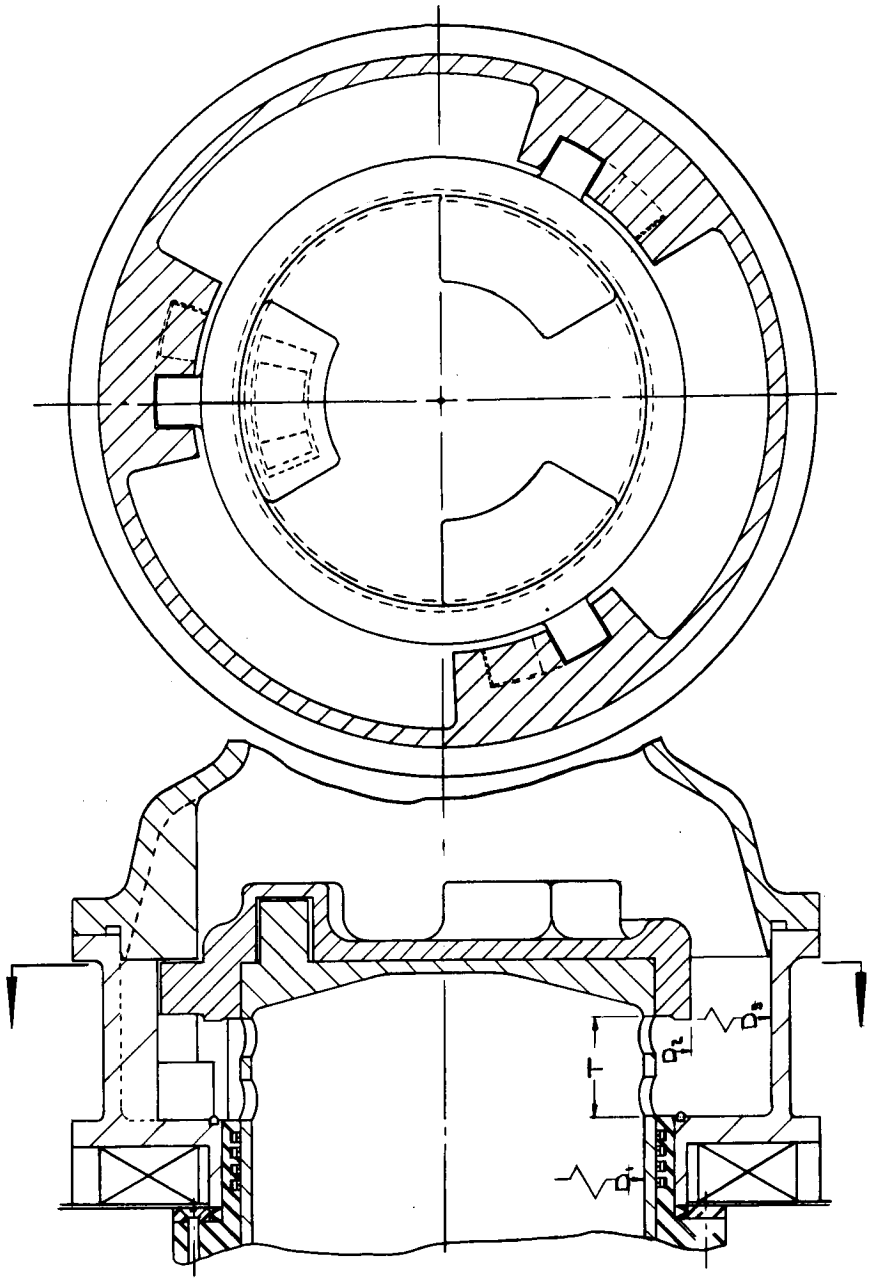
Thrust Versus Duration for Engine Operation on Tank Pressurization Gases, Pressure-Fed System

Figure V-29



Control Moment for 90° Sector by Chamber Pressure Modulation

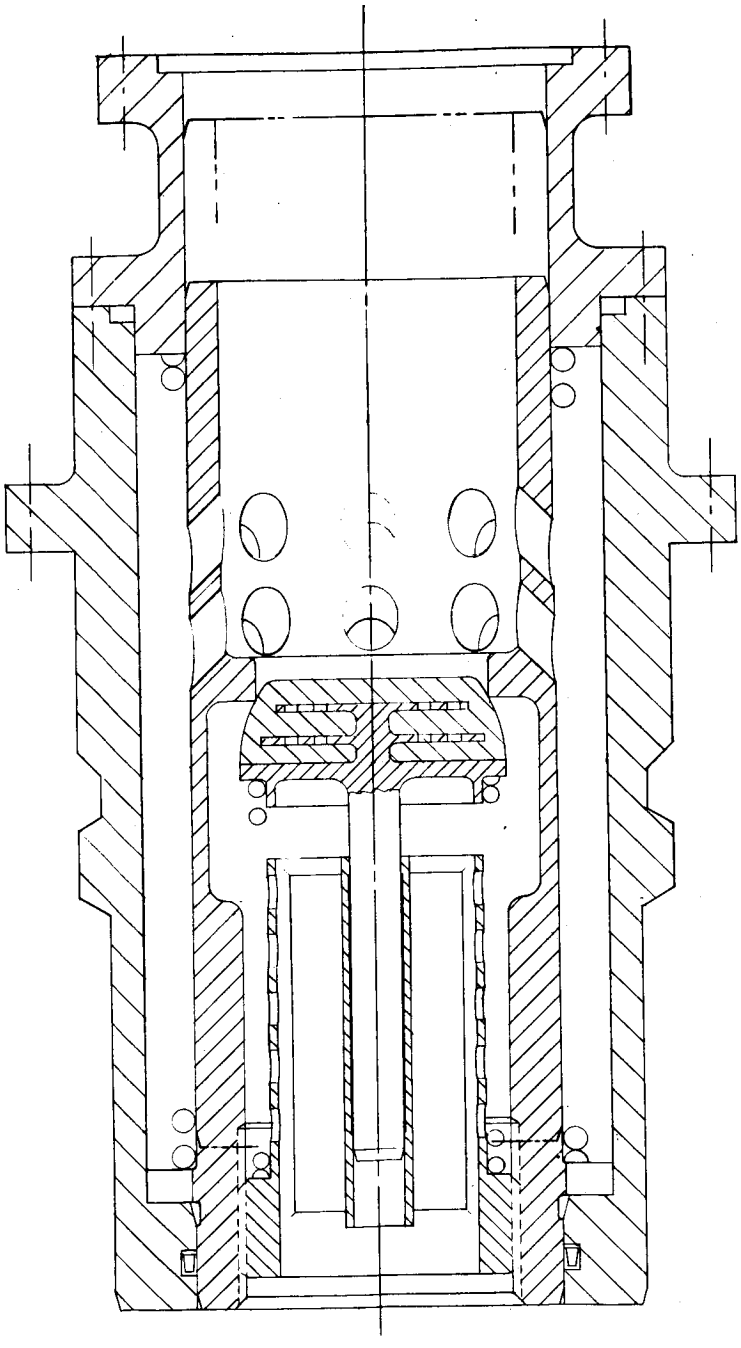
Figure V-30



VALVE SIZE	D_1	D_2	D_3	T
8	8.00	10.00	14.00	3.00
10	10.00	12.00	17.00	3.75
18	18.00	20.00	27.00	8.00

Manually Connected Fill or Vent Valve Coupling

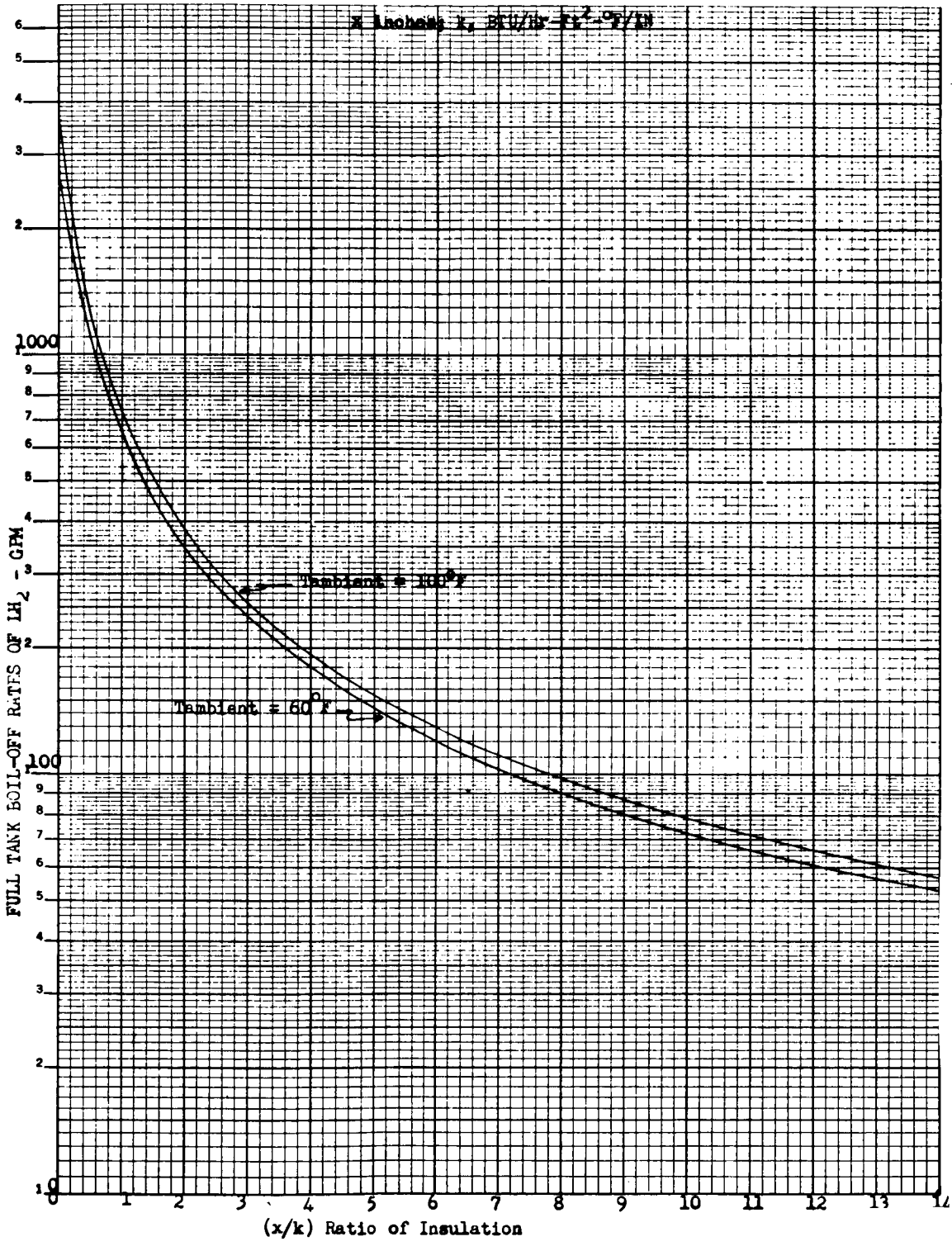
Figure V-31



	OXID VALVE	FUEL VALVE
$\dot{w}$ #/SEC	B	2
WEIGHT	10#	10#

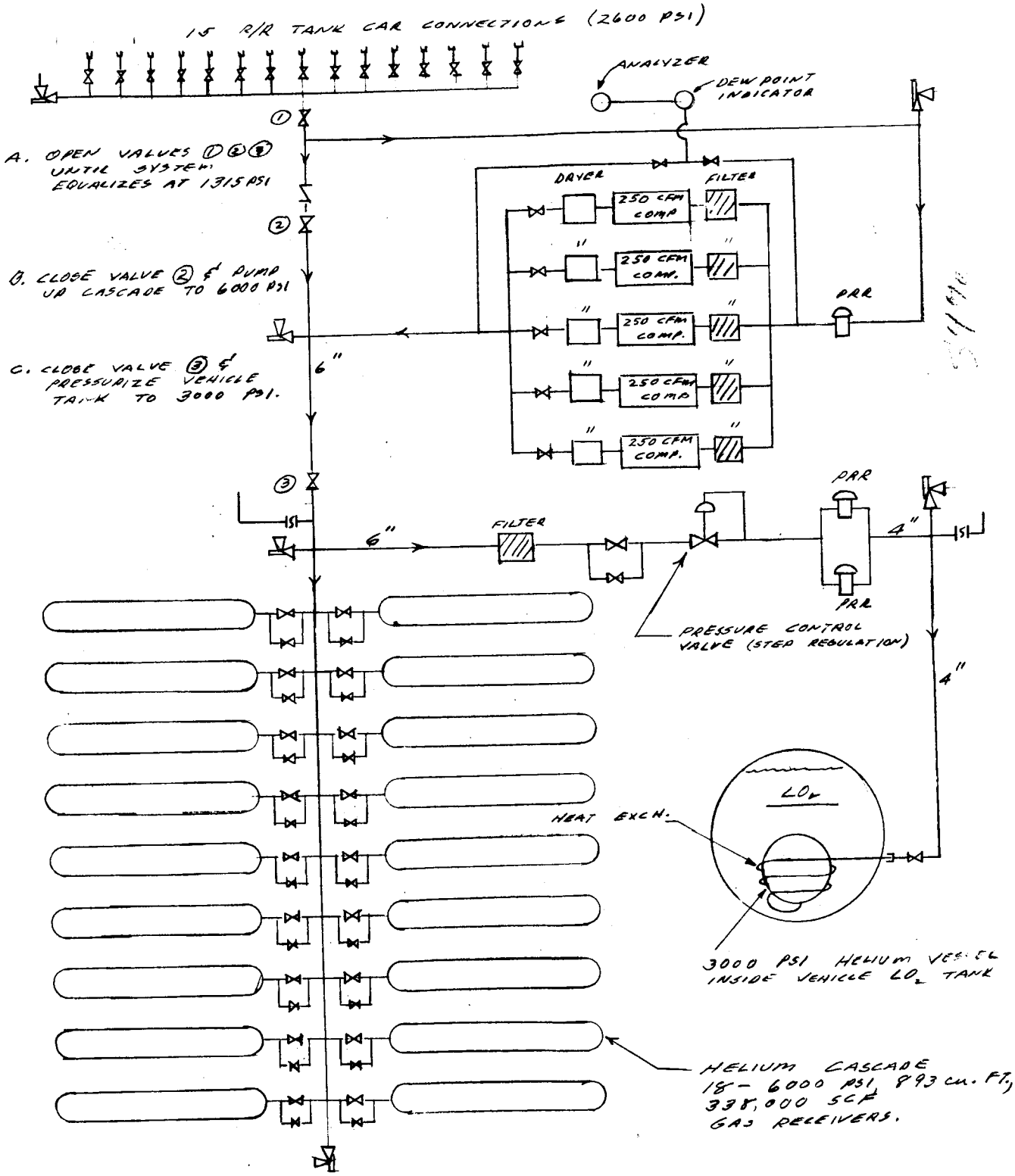
Combination Topping Vent and Pressure Relief Valve

Figure V-32



Primary Hydrogen-Tank Boiloff as Function of Tank Insulation Thickness, Frost Neglected

Figure V-33



Helium Storage System

Figure V-34

VI. INTEGRATION OF BOOSTER SUBSYSTEMS

The results of subsystem design and analysis studies were integrated to provide the overall booster system shown in Figure II-1. A schematic of the system, including major component data, is shown in Figure II-2.

A. SUBSYSTEM CONFIGURATION SELECTION

1. Propellant Tankage

Spherical tanks, fabricated from 2014 T6 aluminum, were used to contain both propellants.

2. Thrust-Chamber Assembly

The thrust-chamber assembly consists of 32 film-cooled chambers operating at 250 psi and a mixture ratio of 6:1.

3. Pressurization System

Studies indicated that no single system was clearly superior. However, to facilitate a cost and weight study, heated helium was selected for oxidizer tank pressurization and heated hydrogen pressurized by heated helium was selected for fuel tank pressurization.

4. Vector Control

Vector control is obtained by chamber pressure modulation. Four 90° sectors are throttled as required to provide the corrective moment.

VI, A, Subsystem Configuration Selection (cont.)

5. Roll Control

Two auxiliary engines, located 180° apart, provide roll control.

6. Power Supply

Power for all control components is provided by a 28 v dc battery.

VI, Integration of Booster Subsystems (cont.)

B. BOOSTER TRANSIENT OPERATION

1. Start Transient

a. Propellant tanks are prepressurized to operating tank pressures. This pressure may be provided by either ground-supplied gas pressure, or by closing the tank vents, allowing boiling propellants to create the required pressure.

b. Thrust-chamber and roll control engine igniters are supplied with power.

c. The helium burst diaphragm valve is electrically actuated, admitting helium to the inlet of the main pressure regulator. The regulator begins to operate, governing pressurization gases to the main propellant tanks. As the pressure rises, the burst diaphragms at the inlet to the oxidizer and auxiliary hydrogen tanks are ruptured, allowing gases to enter the tanks.

d. Thrust-chamber valves are opened, admitting propellants to the thrust-chamber assembly. As the pressure rises, burst diaphragms in the propellant lines to the roll control chambers are ruptured, admitting propellants to the roll control chambers. The propellants in all chambers burn to provide the required thrust. It is anticipated that after initial burning of the propellants, thrust will decay temporarily until the pressurization system becomes self-sustaining.

e. Power to the igniters is terminated and the engine reaches rated thrust.

VI, B, Booster Transient Operation (cont.)

2. Steady-State Operation

a. Vector control signals from the guidance system are interpreted and transmitted to the thrust-chamber valve actuators, which, in turn, position the thrust-chamber valves to provide the chamber pressure reduction required to effect the desired correction. Control is obtained by throttling chamber pressure in 90° sectors.

b. Roll control signals from the guidance system are interpreted and transmitted to the roll control actuators, which then operate the roll control engines into equal but opposite positions, creating a force couple to obtain the desired corrective control moment.

3. Shutdown Transient

a. The helium burst diaphragm valve is closed, preventing further pressurization of the propellant tanks.

b. The thrust chamber valves are then closed, terminating propellant flow to the thrust chamber assembly and the roll control engines.

VI, Integration of Booster Subsystems (cont.)

C. PROPELLANT UTILIZATION CONSIDERATIONS

Propellant utilization is a control placed in vehicle systems to ensure simultaneous, or near simultaneous, exhaustion of propellants in the propellant tank at engine burnout. This control is normally effected by periodically, or continually, changing engine mixture ratio so that the mean engine mixture ratio throughout the run is equal to the propellant tank loading ratio. Without control, these mixture ratios are different because of propellant loading inaccuracies, engine mixture ratio shift resulting from engine tolerances, and engine mixture ratio shift caused by changing static and acceleration heads.

In engine systems without propellant utilization provisions, propellant outage weight can be minimized by proper selection of the propellant tank loading mixture ratio. Knowing the static and acceleration heads during flight, the mean engine mixture ratio (excluding engine tolerance effects) can be calculated. The mean mixture ratio is then modified to include engine tolerance effects and loading inaccuracies (the procedure for which is discussed below), and the propellant tanks are loaded to this mixture ratio.

In propulsion systems in which engine mixture ratio is high and the fuel density is low(as is the case with LO_2/LH_2 systems), minimizing residual propellant weight by loading extra fuel to ensure oxidizer exhaustion is attractive. Although a detailed analysis of engine tolerances was not made during this study, results from other studies indicate that loading 2% extra fuel (which is less than 0.5% of total propellants) will ensure oxidizer exhaustion. On this basis, it was decided not to incorporate a propellant utilization system in the booster design. It is emphasized that a detailed analysis of this subject was not made, and it is possible that the incorporation of propellant utilization may represent a lighter system. However, the potential system reliability would be significantly increased if a propellant utilization system is not used.

VI, Integration of Booster Subsystems (cont.)

D. BOOSTER PRESSURE SCHEDULE

System pressure drops were calculated for both the main propellant systems and the pressurization systems, and are summarized in Tables VI-1 and VI-2. Final analysis indicates that the required oxidizer and fuel tank pressures required are 272 psia and 312 psia, as opposed to the 265 psia and 310 psia assumed during the study. Because the difference is small and the accuracy of the component pressure drops calculated are not within this order, the booster design was not changed from the original assumption.

E. WEIGHT SUMMARY

All components were designed to the point where component weight could reasonably be estimated. A weight summary including all booster components is given in Table VI-3. Total booster takeoff weight is 3,491,615 lb. Inert weight is 558,175 lb. Stage mass fraction was calculated to be 0.840.

F. FINAL TRAJECTORY COMPUTATION AND BOOSTER PERFORMANCE

At the conclusion of the study program, a final trajectory analysis was made with actual vehicle weights and specific impulses. Initial estimates were given in Table IV-1. The resulting final vehicle performance is presented in Table VI-4. Booster stage payload is 1,868,385 lb and payload-to-orbit with the assumed second stage is 283,810 lb.

(End text)

TABLE VI-1

BOOSTER PRESSURE SCHEDULE--MAIN PROPELLANT SYSTEM

PRESSURE	OXIDIZER	FUEL
Thrust Chamber (injector face)	263	
ΔP injector	45	35
Injector Inlet	308	298
Δ P Manifold	5	4
Manifold Inlet	313	302
ΔP TCV	5	3
TCV Inlet	318	305
ΔP Line and orifice	8	7
TCA Inlet Pressure	326	312
Limiting Acceleration Head Pressure	52	-1
Maximum Propellant Tank Pressure*	274	313

* Propellant Tank Pressures Used for Study were: Oxidizer 265 psia, and fuel 310 psia

TABLE VI-2

BOOSTER PRESSURE SCHEDULE--PRESSURIZATION SYSTEM

PRESSURE	OXIDIZER	FUEL
Propellant tank pressure	265	310
H_2 pressurization line and orifice ΔP from tank to plug heat exchanger (H_2)	-	10
Plug heat exchanger (H_2) outlet manifold outlet pressure	-	320
Plug heat exchanger (H_2) outlet manifold ΔP	-	17
Plug heat exchanger (H_2) outlet pressure	-	337
Plug heat exchanger (H_2) ΔP	-	50
Plug heat exchanger inlet pressure	-	387
H_2 pressurization line ΔP from auxiliary hydrogen bottle to plug heat exchanger (H_2)	-	11
Auxiliary hydrogen bottle pressure	-	398
He pressurization line and orifice ΔP from plug heat exchanger (He) to auxiliary hydrogen bottle	-	10
He pressurization line and orifice ΔP from plug heat exchanger (He) to oxidizer tank	143	-
Plug heat exchanger (He) outlet manifold outlet pressure	408	
Plug heat exchanger (He) outlet manifold ΔP	16	
Plug heat exchanger (He) outlet pressure	424	
Plug heat exchanger (He) ΔP	80	

TABLE VI-2 (cont.)

PRESSURE	OXIDIZER	FUEL
Plug heat exchanger (He) inlet pressure	504	
Plug heat exchanger (He) helium supply line ΔP	10	
Helium pressure regulator outlet pressure	514	
Helium pressure regulator ΔP	52	
Helium pressure regulator inlet pressure	566	
Oxidizer tank helium heat exchanger ΔP	10	
Helium storage bottle, minimum pressure	576	
Helium storage bottle, initial pressure	3000	

TABLE VI-3

SYSTEM WEIGHT SUMMARY--6,700K PRESSURE-FED ENGINE

		COMPONENT DRY WEIGHTS	TOTAL WEIGHTS
1.	**Usable Propellants**		
	Fuel	421,480	
	Oxidizer	2,511,960	2,933,440
2.	**Unusable Propellants (outage)**		
	Fuel	3,000	
	Oxidizer	36,000	39,000
3.	**Propellant Tankage Systems**		
	Fuel Tank	190,000	
	Oxidizer Tank	73,000	
	Fuel Tank Insulation	700	
	Main and Topping Fill Valves	240	
	Main and Topping Vent Valves	240	
	Vapor Isolation Burst Diaphragm Valve for Fuel Tank	40	264,220
4.	**Airframe Skin and Structure**	50,000	50,000
5.	**Main Propellant Lines**		
	Fuel	357	
	Oxidizer	1,876	2,233

TABLE VI-3 (cont.)

		COMPONENT DRY WEIGHTS	TOTAL WEIGHTS
6.	**Thrust Chamber Assembly**		
	Chambers	8,163	
	Injectors	8,020	
	Plug	15,686	
	Plug Heat Exchangers and Manifolds	5,009	
	Thrust Chamber Valves and Actuators	3,760	
	Fuel Manifold	655	
	Oxidizer Manifold	655	
	Thrust chamber mounting ring and tank reinforcement	10,500	
	Thrust chamber lines	109	
			52,557
7.	**Pressurization System**		
	Pressurant Gas: Hydrogen	25,760	
	Pressurant Gas: Helium	25,100	
	Auxiliary hydrogen tank and insulation	11,105	
	Helium Tank	78,800	
	Helium heat exchanger in oxidizer tank	2,270	
	Helium Regulator	75	
	Fill and Vent Valves for pressurant tanks	50	
	Helium Safety Shuroff valve-burst diaphragm type	20	
	Vapor isolation burst diaphragm valve for auxiliary hydrogen tank	10	
	Fuel tank pressurization lines	510	
	Oxidizer Tank Pressurization lines	615	
	Pressurization line insulation	300	
			144,615

TABLE VI-3 (cont.)

		COMPONENT DRY WEIGHTS	TOTAL WEIGHTS
8.	Roll Control Engines		
	Thrust Chambers	400	
	Actuators	50	
	Propellant Lines	30	480
9.	Miscellaneous		
	Sequence Unit	20	
	Battery Power Supply	50	
	Miscellaneous Hardware	5,000	
			5,070
TOTAL BOOSTER WEIGHT			3,491,615

Stage Mass Ratio = 0.84014

TABLE VI-4

300-NM ORBIT PERFORMANCE FOR A PRESSURE-FED BOOSTER WITH PUMP-FED SECOND STAGE (O_2/H_2 PROPELLANTS)

PARAMETER	1st STAGE(1)	2nd STAGE(2)
Thrust		
Sea Level	6.7×10^6	---
Vacuum	8.65×10^6	3.21×10^6
Engine Specific Impulse		
Sea Level	302	---
Vacuum	390	426
Lightoff Weight - lbs	5,360,000	1,868,385
Useable Propellant Wt - lbs	2,933,440	1,473,293
Burnout Weight - lbs(3)	2,426,560	397,092
Jettison Weight - lbs(4)	558,175	113,382
Payload - lbs	---	283,810
Engine Propellant Weight Flow Rate (lbs/sec)	22,185	7,544
Duration - Secs	132.2	195
Ideal Velocity (ft/sec)	9,944	21,216
Useful Propellant Weight Ratio	.840	.929

(1) Plug Nozzle, Pc = 250 psia, A_e/A_t = 18/1, MR = 6.0(92% of theoretical shift)

(2) DeLaval Nozzle, Pc = 1000 psia, A_e/A_t = 40/1, MR = 6.0 (94% of theoretical shift)

(3) Burnout Weight includes Stage Inert Weight, Propellant Outage of Upper Stages and Payload

(4) Jettison Weight Includes Stage Inert Weight and Propellant Outage

APPENDIX I

PROPELLANT PERFORMANCE AND PROPERTIES

TABLE OF CONTENTS

		PAGE NO.
I.	INTRODUCTION	I-1
II.	DISCUSSION	II-1
	A. THEORETICAL PROPELLANT PERFORMANCE	II-2
	B. PROPERTIES OF PROPELLANTS	II-13
III.	ENGINE DESIGN HANDBOOK DATA	III-1
	A. EXPLANATION OF ENGINE DESIGN HANDBOOK FORMAT	III-1

TABLE LIST

	TABLE NO.
Theoretical Performance and Physical Characteristics of LOX/LH_2 Gas Generators	II-1
Theoretical Performance and Physical Characteristics of Fuel-Rich LOX/C_3H_8 Gas Generators	II-2
Theoretical Performance and Physical Characteristics of Fuel-Rich LOX/RP-1 Gas Generators	II-3
Theoretical Performance and Physical Characteristics of Oxidizer-Rich LOX/RP-1 Gas Generators	II-4
Theoretical Performance and Physical Characteristics of $N_2O_4/.5N_2H_4$ + 0.5 UDMH Gas Generators	II-5
Theoretical Performance and Physical Characteristics of Fuel Rich $N_2O_4/.5N_2H_4$ + .4875 UDMH + .0175 H_2O Gas Generators	II-6
Theoretical Performance of Various Propellants	II-7
Physical Properties of Oxygen and Light Hydrocarbons	II-8

TABLE LIST (cont.)

	TABLE NO.
LOX/LH_2, P_c = 50	III-1
LOX/LH_2, P_c = 100	III-2
LOX/LH_2, P_c = 300	III-3
LOX/LH_2, P_c = 500	III-4
LOX/LH_2, P_c = 700	III-5
LOX/LH_2, P_c = 1,000	III-6
LOX/LH_2, P_c = 2,000	III-7
LOX/LH_2, P_c = 4,000	III-8
LOX/LH_2, P_c = 6,000	III-9
LOX/LH_2, P_c = 10,000	III-10
LOX/C_3H_8, P_c = 300	III-11
LOX/C_3H_8, P_c = 500	III-12
LOX/C_3H_8, P_c = 1,000	III-13
LOX/C_3H_8, P_c = 2,000	III-14
LOX/RP-1, P_c = 300	III-15
LOX/RP-1, P_c = 400	III-16
LOX/RP-1, P_c = 500	III-17
LOX/RP-1, P_c = 600	III-18
LOX/RP-1, P_c = 700	III-19
LOX/RP-1, P_c = 800	III-20

TABLE LIST (cont.)

	TABLE NO.
LOX/RP-1, P_c = 900	III-21
LOX/RP-1, P_c = 1,000	III-22
LOX/RP-1, P_c = 3,000	III-23
LOX/RP-1, P_c = 5,000	III-24
LOX/RP-1, P_c = 10,000	III-25
$N_2O_4/.5N_2H_4 + .5UDMH$, P_c = 50	III-26
$N_2O_4/.5N_2H_4 + .5UDMH$, P_c = 300	III-27
$N_2O_4/.5N_2H_4 + .5UDMH$, P_c = 500	III-28
$N_2O_4/.5N_2H_4 + .5UDMH$, P_c = 1,000	III-29

FIGURE LIST

	FIGURE NO.
Theoretical Performance and Physical Characteristics of Fuel-Rich Gas Generator -- LOX/LH_2	II-1
Theoretical Performance of LOX (n. b. p.)/Hydrocarbons of Mixture Ratios Which Yield Maximum Specific Impulse	II-2
Heats of Formation of Hydrocarbons	II-3
Heat of Formation Corrections	II-4
Specific Impulses and Combustion Temperatures of LOX + LN_2/RP-1, CH_4, and LH_2	II-5

I. INTRODUCTION

This investigation was for the purpose of providing propellant performance and property information to assist in determining the most feasible large thrust chamber assemblies for specific applications. Also, it provides necessary theoretical performance and physical property data for vehicle analyses as well as studies of controls and rotating machinery. However, because of the complex and varied effects of the propellants upon missile system design and development, it was beyond the scope of this effort to make specific determinations as to which propellants should be used to obtain vehicle designs that provide minimum cost-per-pound of payload.

Propellants considered included selected cryogenics, partial cryogenics, and storables, but excluded all halogen-containing propellants because of cost, availability, handling, and toxicity considerations. Theoretical performance is given for both thrust chambers and gas generators over a range of mixture ratios, chamber pressures, and area ratios. Several propellant systems were considered. The references to the physical properties of propellants cover the state properties, the temperature, and pressure dependent properties.

II. DISCUSSION

Specific propellants cannot be selected for a vehicle of minimum cost-per-pound of payload without considering the effects that propellants have upon the design and development of various missile system components. Conversely, these effects cannot be determined or considered prior to propellant selection. Because specific propellants cannot be selected at this time, a parametric approach, tempered by experience, has been taken.

Experience shows that propellants selected for any missile system possess at least two of the following desirable characteristics: high specific impulse or high bulk density; ease in handling and use; low toxicity, low cost; and availability. Considerations, such as cost, availability, ease of handling and use, as well as toxicity has led to the conclusion that halogen-containing propellants are not suitable for very large booster applications. Unfortunately, similar considerations do not allow other classes of propellants to be rejected as summarily.

Propellants studied include cryogenic, partial cryogenic, and storable systems having a wide latitude of characteristics. The propellants LOX/LH_2, LOX/LC_3H_8 (propane), LOX/RP-1, and N_2O_4/Aerozine-50 were intuitively selected as the best propellants from their respective classes and comprehensively studied.

More limited studies were conducted of the propellants LOX/hydrocarbons (general), LOX + LN_2/LH_2, LOX + LN_2/LCH_4 (methane), and LOX + LN_2/RP-1. These studies included theoretical performance as well as the referencing of physical properties.

II, Discussion (cont.)

A. THEORETICAL PROPELLANT PERFORMANCE

1. Detailed Study of Cryogenics

The only cryogenic propellant system studied in detail is LOX/LH_2. This system was selected on the basis of its high specific impulse, ease in handling and use, low toxicity, low cost, and good availability relative to other cryogenics.

a. Liquid Oxygen/Liquid Hydrogen

Theoretical performance of LOX/LH_2 was computed for a wide range of operating conditions using the general computer program to calculate rocket-engine performance parameters and chemical composition. The range of operating conditions studied includes: chamber pressures from 50 to 10,000 psia; mixture ratios from 0.5 to 20; and pressure ratios to 1,000. These computations provide a library of basic performance and chemical composition data in the form of punched IBM cards, magnetic tapes, print-out sheets. Data covering the ranges shown below are available in the form shown in Section III of this appendix. Data for other conditions is in less available form.

(1) Chamber pressures: 50, 100, 300, 500, 700, 1,000, 2,000, 4,000, 6,000, and 10,000 psia

(2) Mixture ratios: 2 to 10 (max), 3.0 to 6.5 (min)

(3) Area ratios: 1, 2, 3, 4, 5, 6, 7, 8, 10, 12, 15, 20, 30, 40, and 50

II, A, Theoretical Propellant Performance (cont.)

(4) Altitudes: sea-level and infinite (vacuum)

The information tabulated at each pressure, mixture ratio, area ratio point includes: characteristic exhaust velocity (c*); exhaust velocity (V_e) in relationship to characteristic exhaust velocity; absolute exit pressure, nozzle exit area ($P_e A_e$) in relationship to absolute chamber pressure, throat area ($P_c A_t$); thrust coefficient (C_f), vacuum; thrust coefficient, sea-level; combustion temperature (T_c); and exit temperature (T_e). This data is presented in Tables III-1 through III-10 of this appendix for the respective chamber pressures. Similar performance data can be easily obtained for other area ratios and altitudes if necessary. An explanation of the handbook data format is presented as a preface to Section III, this appendix.

Currently, chemical compositions of combustion gases have little use in thrust chamber design except as input data for heat transfer problems. Because this data is voluminous and most easily used in heat transfer programs as magnetic tape input, they are not included as a part of this report.

Theoretical performance and physical characteristics of a fuel-rich LOX/LH_2 gas generator were computed and are shown in Figure II-1. Similar data was tabulated for both fuel-rich and oxidizer-rich gas generators (Table II-1).

2. Detailed Study of Partial Cryogenics

Selection of the propellant combinations LOX/LC_3H_8 (propane) and LOX/RP-1 for detailed investigation was based upon the same criteria as the choice of cryogenics, and from information gained during general studies of LOX/hydrocarbons (Section II, A, 4, this appendix).

II, A, Theoretical Propellant Performance (cont.)

a. Liquid Oxygen/Liquid Propane

The theoretical performance of LOX/LC_3H_8 was computed and the various data processed in a manner similar to that described for LOX/H_2 (Section II, A, 1, b, this appendix). The range of chamber pressures studied for LOX/LC_3H_8 was narrowed (300 to 2,000 psia) because of results obtained from preliminary design studies of very high pressure LOX/H_2 thrust chamber assemblies. A handbook of performance data for LOX/LC_3H_8 is given in Tables III-11 through III-14 of this appendix for the following conditions:

(1) Chamber pressures: 300, 500, 1,000, and 2,000 psia

(2) Mixture ratios: 2.0, 2.4, 2.8, 3.2, 3.6, and 4.0

(3) Area ratios: 1, 2, 3, 4, 5, 6, 7, 8, 10, 12, 15, 20, 30, 40, and 50

(4) Altitudes: sea-level and infinite (vacuum)

See Section II, A, 1 and the preface to Section III, this appendix for an explanation of the data.

Theoretical performance and physical characteristics of fuel-rich LOX/LC_3H_8 gas generators were computed and are given in Table II-2.

The chemical compositions of gas generator and thrust chamber combustion gases are not included in this report; however, they are available in both printed form and on magnetic tapes.

II, A, Theoretical Propellant Performance (cont.)

b. Liquid Oxygen/RP-1

A large amount of performance and chemical composition data has been computed and accumulated for LOX/RP-1 since it was first studied at Aerojet-General in some detail late in 1957. The basic performance data available on IBM card and from magnetic tapes was processed to provide a handbook of performance data. This handbook is given in Tables III-15 through III-25 of this appendix for the following conditions:

(1) Chamber pressures: 300, 400, 500, 600, 700, 800, 900, 1, 000, 3, 000, 5, 000, and 10, 000 psia

(2) Mixture ratios: 1.8 to 3.2 or 3.4

(3) Area ratios: 1, 2, 3, 4, 5, 6, 7, 8, 10, 12, 15, 20, 30, 40, and 50

(4) Altitudes: sea-level and infinite (vacuum)

Section II, A, 1 and the preface to Section III, this appendix explains this data. Inconsistencies in the data amounting to approximately 0.1% are known to occur between the old and most recent data. These inconsistencies result from a combination of program changes and variations in the heats of formation of RP-1 used for input. The variations are unimportant, however, because performance of LOX/RP-1 can vary approximately 1.2% as a result of the RP-1 variations allowed by military specification.

Theoretical performance and physical characteristics of fuel-rich and oxidizer-rich LOX/RP-1 gas generators are given in Tables II-3 and II-4, respectively.

II, A, Theoretical Propellant Performance (cont.)

The chemical compositions of gas generator and thrust chamber combustion gases are not included in this report, but are available in printed form and partially available on magnetic tapes.

3. Detailed Study of Storables

a. N_2O_4/Aerozine-50

This propellant system is the highest performing storable system available for large boosters using current technology. Its performance has been studied in detail since 1957. Performance data for this system is summarized in Tables III-26 through III-29 of this appendix for the following conditions:

(1) Chamber pressures: 50, 300, 500, 1000 psia

(2) Mixture ratios: 1.0 to 2.6 (minimum)

(3) Area ratios: 1, 2, 3, 4, 5, 6, 7, 8, 10, 12, 15, 20, 30, 40, and 50

(4) Altitudes: sea-level and infinite (vacuum)

Theoretical performance and physical characteristics of fuel-rich and oxidizer-rich N_2O_4/Aerozine-50 gas generators are given in Tables II-5 and II-6*. In Table II-5, Aerozine-50 is assumed to be 50% w N_2O_4 + 50% w UDMH. In Table II-6, Aerozine-50 is assumed to be 50% w N_2O_4 + 48.75% UDMH + 1.75% w H_2O.

* Chemical compositions of gas generator and thrust chamber combustion gases are available from the Aerojet-General Corporation Liquid Rocket Plant.

II, A, Theoretical Propellant Performance (cont.)

4. Miscellaneous Performance Studies

a. Reliability of Performance Calculations at Very High Chamber Pressures

The reliability of performance calculations at a chamber pressure of 10,000 psia has been briefly investigated for LOX/LH_2 at a mixture ratio of 6:1. Estimates of deviations from ideal gas behavior (a basic assumption in the computations) indicate that the combustion gases have a compressibility factor of approximately 1.02 in the chamber and throat. Thus, the gases may actually occupy approximately 2% more volume than they would ideally. This deviation is considered to be quite insignificant. The nonideality of combustion gases also affects their composition, which, in turn, determines performance. The magnitude of the effect is believed to be small, but cannot be quantitatively determined without a knowledge of the fugacities and activity coefficients of the combustion products. These values are not available.

Deviations of similar magnitude can be expected for all C, H, O_2, N_2 propellant systems and none is considered to be significant at 10,000 psia and the combustion temperatures involved. At lower pressures, the computations become even more reliable.

b. Theoretical Performance of Liquid Oxygen and Hydrocarbons

The number of hydrocarbons that can be considered as potential rocket fuels in combustion with liquid oxygen is enormous; therefore, a generalized approach for determining propellant performance is necessary.

II, A, Theoretical Propellant Performance (cont.)

Theoretical performance of any hydrocarbon, or mixture of hydrocarbons with liquid oxygen, at a given combustion and exhaust conditions, is dependent upon the following variables:

(1) Carbon/hydrogen ratio of the fuel

(2) Heat of formation of the fuel (which is, in turn, dependent upon molecular structure and physical condition)

(3) Ratio of oxidizer to fuel

(4) Physical condition of the oxidizer

For this investigation, a chamber pressure of 1,000 psia with optimum expansion to atmospheric pressure and complete shifting chemical equilibrium was selected as the primary theoretical combustion model. Because theoretical performance can be conveniently represented graphically as a function of only two variables, the two other variables must be stipulated. The carbon/hydrogen ratio of the fuel and heat of formation of the fuel are considered to be primary variables, while the physical condition of the oxidizer is stipulated to be liquid at its normal boiling point (-297°F). The ratio of oxidizer to fuel (mixture ratio) is stipulated to be that mixture ratio which yields maximum specific impulse.

The performance of liquid oxygen/hydrocarbons was computed and the indicated criteria observed. Results of this investigation are summarized in Figure II-2, where maximum specific impulses as well as mixture ratios and combustion temperatures at maximum specific impulse are shown as functions of the carbon/hydrogen molar ratio and heat of formation of the fuel.

II, A, Theoretical Propellant Performance (cont.)

Heats of formation of approximately 80 hydrocarbons of seven different families of compounds are shown in Figure II-3 as a function of the number of carbon atoms per molecule. Names of the families of compounds, their chemical formulas, and their carbon/hydrogen molar ratios are also tabulated. Heats of formation of other pure hydrocarbons are available from many chemistry and chemical engineering sources. Heats of formation of petroleum fractions can be readily calculated from heats of combustion and empirical chemical formulas. These can, in turn, be estimated from physical property data, if necessary[1]. Note that the heats of formation of the fuel are not necessarily standard state values but can include energy changes needed to go from standard state to any given condition of physical state, temperature, and pressure. The standard state heat of formation is defined here as the enthalpy of the product minus the standard state enthalpies of the elements from which it is formed. Enthalpies of the elements in their normal physical state and stable allotropic form at 298.15°K and one atmosphere pressure are taken as zero.

The performance of liquid oxygen/hydrocarbon systems in which the liquid oxygen is not at its normal boiling points can also be determined from Figure II-2 by applying a correction to the heat of formation of the fuel. Necessary corrections are shown in Figure II-4 as functions of optimum mixture ratio and condition of the saturated liquid oxygen. Because the heat of formation corrections are functions of mixture ratio, and the mixture ratios are functions of the heat of formation of the fuel, a trial and error solution is required. The use of Figures II-2, II-3, and II-4 in solving such a problem is shown in the following example:

[1] Hougen and Watson, Chemical Process Principles, Part One, pp. 329-338, Wiley, 1950

II, A, Theoretical Propellant Performance (cont.)

PROBLEM: Determine the maximum specific impulse, the combustion temperature and optimum mixture ratio of the system LOX (saturated liquid at 270°F)/Ethylbenzene (liquid at 77°F).

Ethylbenzene is a n-alkylbenzene containing eight carbon atoms per molecule. From Curve 2 of Figure II-3, it is found that the heat of formation of the fuel is -30 cal/g. From the formula given at the top if Figure II-3 for n-alkylbenzenes, it is found that the C/H molar ratio is 0.8. In Figure II-2, it is found that the optimum mixture ratio is about 2.3. From Figure II-4, the correction to be added to the heat of formation of the fuel to compensate for the oxidizer not being at its normal boiling point is found to be 140 cal/g. Thus the new heat of formation of the fuel is -30 + 140 = 110 cal/g. Referring back to Figure II-2, the proper mixture ratio is found to be more nearly 2.27 and from Figure II-4, the proper correction is found to be more nearly 138 cal/g. The solution to the problem is now found in Figure II-2, where the C/H ratio equals 0.8 and the heat of formation of the fuel equals 108 cal/g (-30 + 138).

SOLUTION:	Maximum specific impulse	297.4 sec
	Combustion temperature	6,817°R
	Mixture ratio, O/F	2.27 lb/lb

It can been seen from Figure II-3 that, as hydrocarbon molecules of several types become large, the carbon/hydrogen ratios approach approximately -430 cal/g. This is true of any large hydrocarbon molecule which is dominantly aliphatic or napthenic. Thus, from Figure II-2 it is found that large hydrocarbon molecules that are largely saturated yield approximately 300 sec of specific impulse. RP-1 is a typical example (C/H = 0.476 to 0.502; net heat of combustion = 18,700 $\pm$ 200 Btu/lb; I_{std} = 298.5 to 302.3). Aromatic hydrocarbons

II, A, Theoretical Propellant Performance (cont.)

are characterized by resonance-stabilized structures and relatively high C/H ratios. Both of these characteristics tend to decrease their performance potential. One of the best aromatic fuels, benzene, yields 2 to 3% less specific impulse than RP-1. It is concluded from the foregoing discussion that aromatic hydrocarbons are poorer than RP-1 and paraffins, olefins, naphthenes, and acetylenes with 10 or more carbon atoms per molecule yield little, or no better performance than RP-1. Further study shows that the light paraffins, olefins, acetylenes, and cycloalkanes can yield significant specific impulse increases over RP-1. However, the light acetylenes are shock sensitive while the heavier acetylenes and the cycloalkanes show no performance advantage over the cheaper and more stable paraffins and olefins. The theoretical performance of methane, ethane, propane, isobutane, n-butane, ethylene, propylene, and 1-butene with liquid oxygen is shown in Table II-7 and compared with the systems LOX/LH_2, LOX/RP-1, and N_2O_4/Aerozine-50. Some of the physical properties of oxygen and the hydrocarbons mentioned above are shown in Table II-8.

It is concluded that a propellant system composed of LOX and light paraffins is the most promising hydrocarbon propellant system available for large boosters. Selection of one light paraffin over others (i.e., between methane, ethane, propane, and butane) is difficult because of their similarities in performance, bulk propellant density, and properties. The final choice of the "best" paraffin is likely to be based upon cost and logistics of the various paraffins and is like to be a mixture of several light paraffins. For the purpose of closer evaluation, the performance characteristics of LOX/LC_3H_8 (liquid propane) can be considered characteristic of the "best" paraffin.

II, A, Theoretical Propellant Performance (cont.)

c. Theoretical Performance of Liquid Oxygen/Nitrogen Mixtures with Liquid Hydrogen/Methane and RP-1

The use of LOX/LN_2 mixtures as oxidizers to decrease combustion temperature, thus simplifying thrust chamber design, has been considered with the fuels LH_2, LCH_4, and RP-1. The effect of oxidizer composition upon combustion temperature and specific impulse at a chamber pressure of 1,000 psia and at mixture ratios which are stoichiometric to H_2O, CO, and N_2 is shown in Figure II-5. Although the stoichiometric mixture ratios studied are probably not the most desirable from an over-all performance standpoint, they do clearly show that liquid oxygen must be diluted very substantially with liquid nitrogen to decrease combustion temperatures significantly. Note that oxidizer dilution gives decreases in specific impulse of considerable magnitude.

It is concluded that the use of mixtures of oxygen and nitrogen to reduce combustion temperatures, permitting simplification in thrust chamber design is not worth the corresponding sacrifice in specific impulse. A decrease in combustion temperature can be achieved with a smaller loss in specific impulse by reducing mixture ratio; however, a decrease in bulk density also occurs. When consideration is given to recovering boost vehicles, the use of either gaseous or liquid O_2N_2 mixtures during descent may prove to be desirable because of lower thrust requirements, the availability of O_2N_2 mixtures from the atmosphere, and the low combustion temperatures possible with such mixtures.

II, A, Theoretical Propellant Performance (cont.)

d. Theoretical Performance of Other Propellant Systems

Theoretical performance of 31 common propellant systems has recently been compiled[2]. The performance of many other systems is available[3].

B. PROPERTIES OF PROPELLANTS

The properties of many propellants have been compiled. Several of these compilations are both comprehensive and voluminous. Because the most important of these are readily available, no attempt is made to reproduce this data herein. Samples of this available data as well as sources are indicated.

1. Oxygen

a. The "Liquid Propellants Handbook"[4] offers the following information about the properties of oxygen:

(1) General Identification:

(a) Molecular formula and weight

(b) Transition temperatures (solid oxygen)

2. Performance and Properties of Liquid Propellants, Aerojet-General Corporation Liquid Rocket Plant, Report No. 8160-6S, June 1961
3. In the files of the Research and Materials Department (8160) Engineering and Research Division, Liquid Rocket Plant, Aerojet-General Corporation
4. Liquid Propellants Handbook, Volume 1, Battelle Memorial Institute, Division of Chemical Engineering, Contract NOa(s) 54-597-C

II, B, Properties of Propellants (cont.)

(c) Freezing, melting, boiling, and triple points.

(d) Critical properties.

(2) Phase Properties:

(a) Densities of solid, liquid, and saturated gaseous oxygen

(b) Coefficient of thermal expansion (liquid oxygen)

(c) Coefficients of compressibility

(d) Specific volume and compressibility factor (molecular gaseous oxygen)

(e) Equations of state

(f) Vapor pressure of solid and liquid oxygen

(g) Vapor-liquid equilibrium data for oxygen-nitrogen mixtures at atmospheric pressure.

(h) Solubility of gaseous oxygen in water and ethyl alcohol

(i) Surface tension (liquid oxygen)

II, B, Properties of Propellants (cont.)

(3) Thermodynamic Properties:

(a) Heats of transition (solid oxygen)

(b) Heat of fusion (atmospheric pressure)

(c) Heat capacity of solid, liquid, and gaseous oxygen as well as atomic O

(d) Ratio of heat capacities (gaseous oxygen)

(e) Joule-Thomson coefficient

(f) Temperature-entropy data and diagram

(g) Entropy of atomic O

(h) Enthalpy (liquid and gaseous oxygen as well as atomic O)

(i) Free energy (gaseous oxygen and atomic O)

(j) Heat of disassociation

(k) Equilibrium constants for the disassociation of oxygen to O

(l) Heat of formation (various oxygen species and ions)

II, B, Properties of Propellants (cont.)

(4) Transport Properties:

(a) Viscosity (liquid and gaseous oxygen as well as liquid oxygen-liquid N_2 mixtures)

(b) Thermal conductivity (liquid and gaseous oxygen)

(c) Prandtl number (gaseous oxygen)

(d) Diffusion coefficient (gaseous oxygen in air)

(e) Self-diffusion coefficients (gaseous oxygen)

(f) Velocity of sound (in liquid and gaseous oxygen)

(g) Sonic relaxation time (gaseous oxygen)

(5) Electromagnetic Properties:

(a) Index of refraction (liquid and gaseous oxygen)

(b) Dipole moment (gaseous oxygen)

(c) Dielectric constant (gaseous and liquid oxygen)

(d) Electrical breakdown (liquid oxygen)

II, B, Properties of Propellants(cont.)

(e) Magneto-optic rotation (verdet constants) (gaseous and liquid oxygen as well as liquid oxygen/nitrogen mixtures)

(f) Paramagnetic susceptibility

(6) Analysis, safety, handling, production, and military specifications are also discussed.

b. A number of other references[5, 6, 7] are also available to supplement the indicated oxygen property data.

2. Oxygen-Nitrogen Mixtures

In general, the properties of oxygen-nitrogen mixtures are not compiled, but scattered among the compilations for the properties of oxygen, air, and nitrogen. In many cases, the properties of given mixtures must be estimated based upon the properties of oxygen, air, and nitrogen. References cited for the properties of oxygen also give data for the vapor-liquid equilibrium of mixtures of oxygen and nitrogen.

5. Scott, Russel B., Cryogenic Engineering, D. Van Nostrand Co., Inc., 1959, pp. 81, 268, 270-277
6. AGC, LRP Report No. 8160-6S, op. cit.
7. A Compendum of the Properties of Materials at Low Temperature (Phase 1) Part 1, Properties of Fluids, National Bureau of Standards Cryogenic engineering Laboratory, July, 1960, WADD Technical Report No. 60-56, Part 1, Contract No. AF 33(616)-58-4 Project No. 7360

II, B, Properties of Propellants (cont.)

a. Air

The properties of air are compiled in three readily-available references[8, 9, 10]. This information includes data, such as:

(1) Composition

(2) Molecular weight

(3) Boiling and condensation points.

(4) Critical properties

(5) Density

(6) Coefficients of thermal expansion

(7) Compressibility factors

(8) Equations of state

(9) Vapor pressure

(10) Solubility

8. Liquid Propellants Handbook, Volume 2, Battelle Memorial Institute, Division of Chemical Engineering, Contract NOa(s) 54-597-C
9. Scott, Russell B., op. cit., pp. 284-288
10. WADD Report No. 60-56, op. cit.

II, B, Properties of Propellants (cont.)

(11) Heat capacity

(12) Joule-Thomson coefficients

(13) Entropy

(14) Enthalpy

(15) Thermal conductivity

(16) Viscosity

(17) Prandtl number

b. Nitrogen

A number of sources delineate the properties of nitrogen[11, 12, 13, 14, 15]. The following is a typical example of the information contained in these references:

(1) Molecular formula and weight

(2) Freezing, boiling and triple points

11. Liquid Propellants Handbook, Volume 2, op. cit. (Properties of Pressurizing Gases)
12. Scott, Russell B., op. cit, pp. 277-283
13. AGC, LRP Report No. 8160-6S, op. cit., p. 95
14. Technical Data on Liquid Nitrogen, Linde Air Products Co., Division of Union Carbide and Carbon Corp., 1957
15. WADD Technical Report No. 60-56, op. cit.

II, B, Properties of Propellants (cont.)

(3) Critical properties

(4) Density

(5) Coefficients of thermal expansion

(6) Equations of state

(7) Vapor pressure

(8) Solubility

(9) Heat of fusion and vaporization

(10) Heat capacity

(11) Joule-Thomson coefficient

(12) Entropy

(13) Enthalpy

(14) Viscosity

(15) Thermal conductivity

(16) Prandtl number

(17) Velocity of sound

II, B, Properties of Propellants (cont.)

(18) Expansivity

(19) Specific heat

(20) Phase transition temperatures

(21) Dielectric constant

3. Nitrogen Tetroxide ($N_2O_4 \rightleftharpoons NO_2$)

a. The "Liquid Propellants Handbook"[16] groups the properties of nitrogen tetroxide in a manner similar to that done for oxygen, under the following general headings:

(1) General identification

(2) Phase properties

(3) Thermodynamic properties

(4) Transport properties

(5) Electromagnetic properties

It also discusses analysis, handling, production, and specifications.

16. Liquid Propellants Handbook, Volume 2, op. cit.

II, B, Properties of Propellants (cont.)

b. Three other references are readily available[17, 18, 19], of which the Allied Chemical Bulletin[20] is the most comprehensive.

4. Hydrogen

The properties of hydrogen have received a great deal of attention, which has led to numerous compilations of its properties. However, in most cases the data have not been critically reviewed (particularly for the liquid and supercritical vapor). It is also important to note that hydrogen exists in two different forms as a result of differences in nuclear spins. Many data sources do not differentiate between the forms thus leading to ambiguity. In other cases, data are available only for normal hydrogen while equilibrium hydrogen is of primary interest for rocket engines.

A number of reference works[21, 22, 23, 24, 25] do exist and the last one indicated[26] not only provides a compilation of the properties of hydrogen, but also cites an additional 77 references. The following represents typical data to be found in the indicated sources.

17. AGC, LRP Report No. 8160-6S, op. cit. p. 97
18. Nitrogen Tetroxide Product Bulletin, Allied Chemical Corp., Nitrogen Division
19. Storable Liquid Propellants - Nitrogen Tetroxide/Aerozine-50, Revision C, Aerojet-General Corporation, Liquid Rocket Plant, Report No. LRP 198, June, 1961, Section III.
20. Nitrogen Tetroxide Product Bulletin, op. cit.
21. Liquid Propellants Handbook, Volume 2, op. cit.
22. Scott, Russell B., op. cit., pp. 289-307
23. AGC, LRP Report No. 8160-6S, op. cit.
24. WADD Technical Report No. 60-56, op. cit.
25. Barney, John D. and Magee, Patrick M., The Engineering Properties of Hydrogen, Aerojet-General Corp., Azusa, California, August, 1958.
26. Ibid

II, B, Properties of Propellants (cont.)

a. Ortho-para conversion rate

b. Triple and boiling points

c. Critical constants

d. Thermal conductivity

e. Viscosity

f. Temperature-entropy diagram

g. Vapor pressure

h. Molar volumes

i. Velocity of sound

j. Compressibility coefficients

k. Specific heats

l. Latent heats of fusion

m. Heat of conversion

n. Viscosity

II, B, Properties of Propellants (cont.)

o. Dielectric constant

p. Enthalpy

q. Surface tension

5. Methane (CH_4)

The sources[27,28] of data about the properties of methane provide the following information:

a. Density

b. Thermal conductivity

c. Specific heat

d. Enthalpy

e. Heat of fusion

f. Heat of vaporization

g. Melting curve

---

27. WADD Technical Report No. 60-56, op. cit.
28. Rossini, F.D., et al, Selected Values of Physical and Thermodynamic Properties of Hydrocarbons and Related Compounds, Carnegie Press, Pittsburgh, 1953.

II, B, Properties of Propellants (cont.)

h. Vapor pressure

i. Critical properties

j. Boiling, melting, and triple points

k. Solid transition temperatures

l. Dielectric constant

m. Surface tension

n. Viscosity

o. Thermodynamic properties

6. Propane C_3H_8)

Rossini's text[29] is the best known single source of information about the properties of propane.

7. RP-1

There are only two sources of information[30, 31] readily available for the properties of RP-1. Data includes:

29. Ibid
30. AGC, LRP Report No. 8160-6S, op. cit.
31. Dear, Lee E. and Shurley, L.A., Characteristics of RP-1 Rocket Fuel, Aerojet-General Corporation, Sacramento, California, TCR-70 (Dept 8120), February 1957

II, B, Properties of Propellants (cont.)

a. Distillation curve

b. Specific gravity

c. Viscosity

d. Specific heat

e. Enthalpy

f. Thermal conductivity

g. Prandtl number

h. Heat of combustion

i. Boiling, freezing, and heat transfer characteristics.

8. Aerozine-50

The only two known sources[32, 33] of information about the properties of Aerozine-50 provide the following data:

a. Molecular weight

b. Freezing and boiling points

32. AGC, LRP Report No. 8160-6S, op. cit., p. 59
33. AGC, LRP Report No. 198, op. cit.

II, B, Properties of Propellants (cont.)

c. Critical temperature

d. Critical pressure

e. Heat of formation and vaporization

f. Specific gravity

h. Thermal conductivity

i. Vapor pressure

j. Viscosity

k. Density.

TABLE II-1

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS OF LOX/LH_2 GAS GENERATORS

Mixture Ratio, O/F, lb/lb	c* ft/sec	T_C °F	Molecular Weight	$C_p C_v$	Composition, % by Volume		
					H_2	H_2O	O_2
Fuel Rich GG							
0.5	5,809	498	3.02	1.390	93.70	6.30	-0-
0.6	6,113	699	3.23	1.383	92.44	7.56	-0-
0.8	6,607	1,015	3.63	1.370	89.92	10.08	-0-
1.0	6,986	1,361	4.03	1.356	87.40	12.60	-0-
1.5	7,543	2,138	5.04	1.314	81.10	18.90	-0-
Oxid Rich GG							
62	3,327	2,209	28.825	1.272	-0-	22.69	77.31
78	2,898	1,742	29.416	1.287	-0-	18.47	81.53
94	2,639	1,406	29.822	1.300	-0-	15.57	84.43
110	2,433	1,151	20.117	1.312	-0-	13.46	86.54

TABLE II-2

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS OF FUEL RICH LOX/C_3H_8 GAS GENERATORS

Mixture Ratio, O/F, lb/lb	P_c, psia	c*, ft/sec	T_c, °F	Molecular Weight*	Composition, mole %					
					H_2	H_2O	$C_{(s)}$	CH_4	CO	CO_2
0.25	1,000	----	1,427	12.469	21.75	10.38	34.83	29.17	2.51	1.35
0.35	1,000	----	1,573	12.265	28.09	10.10	31.78	22.11	6.06	1.86
0.45	1,000	----	1,680	12.342	32.80	9.29	27.81	17.56	10.43	2.11
0.55	1,000	----	1,770	12.549	36.68	8.24	23,38	14.26	15.29	2.15
1.09	1,000	----	2,178	14.342	49.89	3.43	-0-	4.47	41.11	1.10

* Mean molecular weight of combustion products including solids.

TABLE II-3

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS OF FUEL RICH LOX/RP-1 GAS GENERATORS

MIXTURE RATIO, O/F, LB/LB	PC, PSIA	C* FT/SEC	TC, °F	MOLECULAR WEIGHT(1)	COMPOSITION, MOLE %					
					H_2	H_2O	$C_{(s)}$	CH_4	CO	CO_2
0.35	200	----	1501	11.371	33.43	6.52	39.74	10.05	8.61	1.65
0.35	500	3618	1607	11.825	30.51	7.03	39.52	12.45	8.83	1.65
0.35	750	3632	1657	12.042	29.12	7.28	39.43	13.60	8.91	1.66
0.35	1000	3641	1692	12.202	28.09	7.47	39.37	14.44	8.95	1.68
0.38	500	----	1639	11.89	31.83	6.75	37.97	11.42	10.35	1.68
0.38	800	----	1698	12.14	30.22	7.05	37.86	12.73	10.44	1.70
0.38	1000	----	1727	12.27	29.43	7.20	37.81	13.31	10.47	1.72
0.40	200	----	1550	11.492	35.48	5.99	37.15	8.52	11.16	1.69
0.40	500	3718	1660	11.938	32.65	6.55	36.92	10.80	11.39	1.69
0.40	750	3733	1711	12.155	31.28	6.81	36.82	11.90	11.47	1.71
0.40	1000	3744	1749	12.315	30.27	7.01	36.75	12.72	11.52	1.73
0.45	500	3808	1708	12.086	34.55	6.03	34.27	9.43	14.04	1.69
0.45	750	3825	1761	12.300	33.21	6.31	34.16	10.48	14.13	1.71
0.45	1000	3837	1800	12.460	32.21	6.52	34.08	11.27	14.19	1.73
0.5	200	----	1637	11.836	38.87	4.87	31.81	6.26	16.58	1.60
0.6	500	----	1840	12.66	39.17	4.44	26.23	6.39	22.32	1.45
0.6	800	----	1906	12.89	37.75	4.81	26.09	7.44	22.41	1.50
0.6	1000	----	1939	13.01	37.03	4.99	26.02	7.98	22.45	1.53

(1) MEAN MOLECULAR WEIGHT OF COMBUSTION PRODUCTS INCLUDING SOLIDS.

CONFIDENTIAL

TABLE II-4

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS
OF OXIDIZER RICH LOX/RP-1 GAS GENERATORS

Mixture Ratio, O/F, lb/lb	c* ft/sec	T_c, °F	Molecular Weight	Composition, % by Volume			
				C_pC_v	CO_2	H_2O	O_2
25.00	----	2,136	31.828	1.270	8.73	8.73	82.54
34.19	2,624	1,500	31.873	1.291	6.46	6.46	87.08
35.00	----	1,457	31.876	1.292	6.31	6.31	87.38
36.13	2,555	1,401	31.880	1.297	6.12	6.12	87.76
38.21	2,485	1,305	31.886	1.300	5.80	5.80	88.40
40.44	2,414	1,212	31.892	1.303	5.49	5.49	89.02
42.81	2,346	1,122	31.898	1.311	5.19	5.19	89.62

TABLE II-5

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS OF $N_2O_4/.5N_2H_4 + .5$ UDMH GAS GENERATORS

MIXTURE RATIO, O/F, LB/LB FUEL TICH G.G.	PC, PSIA	C* FT/SEC	TC, °F	MOLECULAR WEIGHT(1)	CP/CV(2)	COMPOSITION, MOLE %									
						H_2	H_2O	$C_{(s)}$	CH_4	CO	CO_2	N_2	NH_3	O_2	NO
0	300	4222	1547	12.624	1.253	48.74	-0-	4.76	16.24	-0-	-0-	30.14	0.12	-0-	-0-
0	1000	4282	1703	13.633	1.241	44.59	-0-	1.19	21.49	-0-	-0-	32.50	0.22	-0-	-0-
0.06	150	4262	1533	12.068	1.295	52.14	0.91	6.78	10.13	2.00	0.03	27.94	0.07	-0-	-0-
0.06	300	4307	1618	12.529	1.280	50.27	0.97	5.17	12.41	2.05	0.03	28.99	0.10	-0-	-0-
0.06	750	4365	1740	13.237	1.261	47.39	1.06	2.70	15.91	2.14	0.03	30.60	0.17	-0-	-0-
0.08	150	4288	1556	12.081	1.298	52.56	1.06	6.62	9.21	2.74	0.05	27.70	0.06	-0-	-0-
0.08	300	4331	1639	12.542	1.284	50.69	1.15	5.03	11.45	2.79	0.05	28.74	0.09	-0-	-0-
0.08	750	4387	1761	13.243	1.264	47.85	1.27	2.60	14.86	2.90	0.05	30.32	0.16	-0-	-0-
0.10	150	4311	1577	12.116	1.301	52.93	1.17	6.35	8.42	3.49	0.07	27.51	0.06	-0-	-0-
0.10	300	4353	1661	12.562	1.287	51.12	1.28	4.82	10.56	3.56	0.07	28.51	0.09	-0-	-0-
0.10	750	----	1784	13.248	1.268	48.34	1.42	2.46	13.84	3.68	0.07	30.04	0.15	-0-	-0-
0.10	1000	4438	1824	13.496	1.257	47.34	1.48	1.61	15.01	3.72	0.07	30.59	0.18	-0-	-0-
0.20	300	4443	1763	12.817	1.289	52.87	1.45	2.85	7.22	7.57	0.13	27.84	0.07	-0-	-0-
0.20	1000	4527	1926	13.679	1.275	49.38	1.83	- 0 -	11.04	7.78	0.15	29.67	0.15	-0-	-0-
OXID. RICH G.G.															
15.00	200	3004	1900	29.346	1.288	- 0 -	11.83	- 0 -	- 0 -	- 0 -	3.05	34.26	- 0 -	50.81	0.05
22.00	200	2534	1269	29.738	1.314	- 0 -	8.34	- 0 -	- 0 -	- 0 -	2.15	34.00	- 0 -	55.50	- 0 -

(1) MEAN MOLECULAR WEIGHT OF COMBUSTION PRODUCTS INCLUDING SOLIDS.

(2) RATIO OF SPECIFIC HEATS OF GASEOUS COMBUSTION PRODUCTS ONLY.

CONFIDENTIAL

TABLE II-6

THEORETICAL PERFORMANCE AND PHYSICAL CHARACTERISTICS OF FUEL RICH $N_2O_4/.5N_2H_4$ + .4875 UDMH + .0175 H_2O GAS GENERATORS

MIXTURE RATIO, O/F, LB/LB FUEL RICH G.G.	PC, PSIA	C* FT/SEC	Tc, °F	MOLECULAR WEIGHT (1)	CP/CV (2)	COMPOSITION, MOLE %							
						H_2	H_2O	$C_{(s)}$	CH_4	CO	CO_2	N_2	NH_3
0.025	300	4217	1546	12.946	1.282	48.39	1.08	3.01	15.78	1.46	0.03	30.13	0.12
0.025	356	4232	1570	13.058	1.280	47.95	1.08	2.60	16.34	1.48	0.03	30.39	0.13
0.025	400	4236	1583	13.161	1.276	47.53	1.09	2.24	16.86	1.49	0.03	30.62	0.13
0.025	450	4244	1598	13.259	1.271	47.13	1.10	1.89	17.35	1.50	0.03	30.85	0.14
0.05	300	4251	1575	12.901	1.291	49.10	1.42	3.18	14.16	2.33	0.06	29.65	0.11
0.05	356	4263	1597	13.026	1.287	48.60	1.43	2.73	14.77	2.35	0.06	29.93	0.12
0.05	400	4270	1611	13.122	1.282	48.21	1.45	2.40	15.25	2.37	0.06	30.15	0.13
0.05	450	4277	1627	13.214	1.279	47.84	1.46	2.07	15.70	2.38	0.06	30.35	0.13
0.085	300	4295	1613	12.889	1.295	49.99	1.72	3.09	12.26	3.63	0.10	29.12	0.10
0.085	356	4307	1635	13.013	1.291	49.49	1.74	2.65	12.85	3.66	0.10	29.39	0.11
0.085	400	4314	1651	13.100	1.287	49.15	1.76	2.34	13.27	3.68	0.10	29.59	0.12
0.085	450	4322	1666	13.187	1.282	48.80	1.78	2.03	13.69	3.70	0.10	29.78	0.12
0.10	300	4313	1629	12.899	1.300	50.34	1.80	2.95	11.55	4.21	0.12	29.94	0.09
0.10	356	4323	1651	13.022	1.295	49.85	1.83	2.52	12.13	4.24	0.12	29.21	0.10
0.10	400	4332	1667	13.108	1.290	49.51	1.85	2.22	12.54	4.26	0.12	29.40	0.11
0.10	450	4339	1682	13.197	1.286	49.15	1.87	1.91	12.96	4.28	0.12	29.59	0.12

(1) MEAN MOLECULAR WEIGHT OF COMBUSTION PRODUCTS INCLUDING SOLIDS.

(2) RATIO OF SPECIFIC HEATS OF GASEOUS COMBUSTION PRODUCTS ONLY.

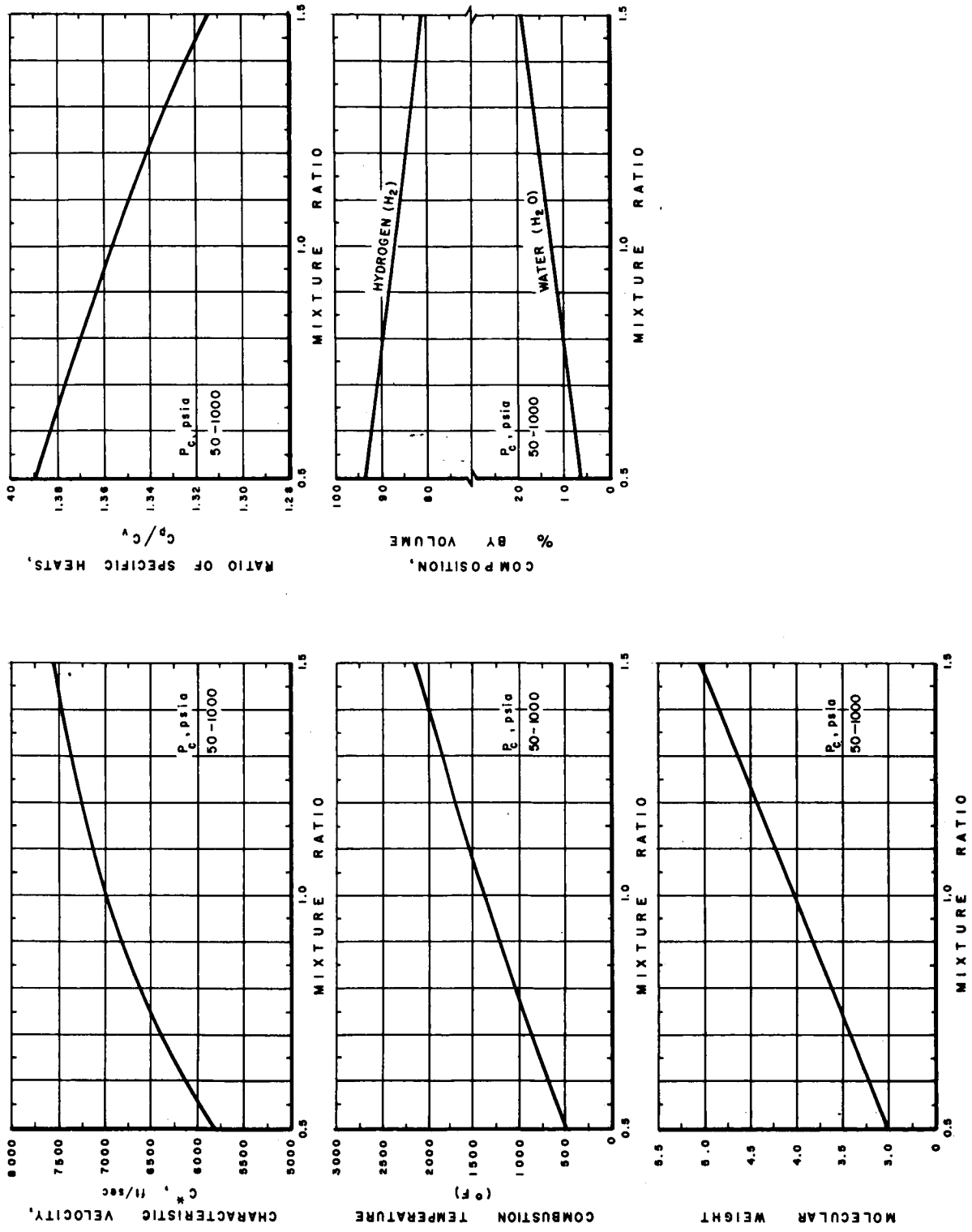
CONFIDENTIAL

TABLE II-7

THEORETICAL PERFORMANCE OF VARIOUS PROPELLANTS

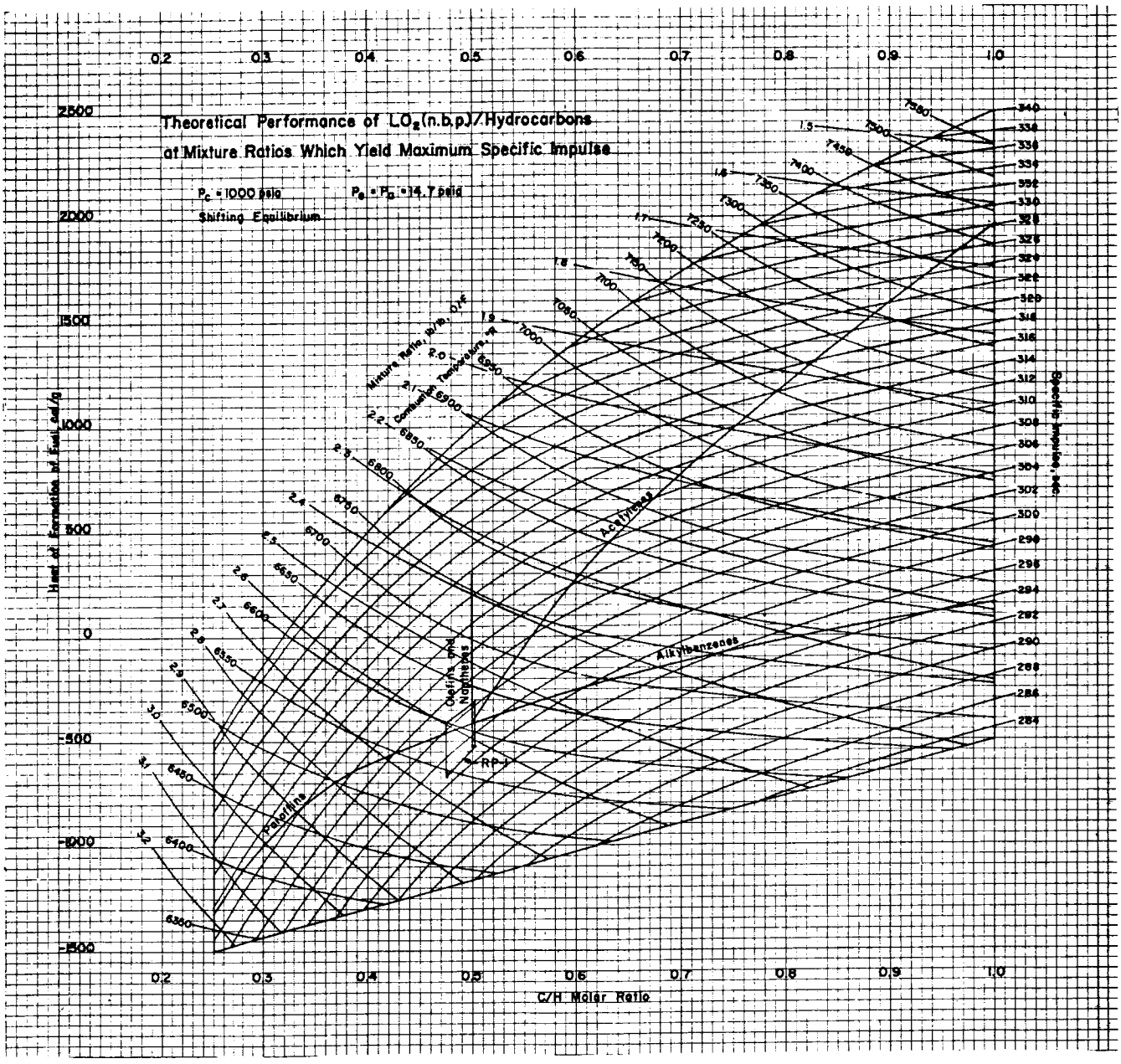
P_c = 1,000 psia $P_e = P_a$ = 14.7 psia SHIFTING EQUILIBRIUM

System	I_{std}	MR	T_c, °R	Bulk Density	$I_{std} \times D$
LOX(NBP)/LH_2(NBP)	383.8	6	6,339	22.54	8,651
LOX(NBP)/LCH_4(NBP)	309.3	3.19	6,359	50.77	15,703
LOX(NBP)/LC_2H_6(NBP)	308.2	2.90	6,462	55.72	17,173
LOX(NBP)/LC_3H_8(NBP)	306.4	2.79	6,502	56.81	17,407
LOX(NBP)/L ISO-C_4H_{10}(77°F)	305.0	2.74	6,523	55.41	16,900
LOX(NBP)/L_N-C_4G_{10}(77°F)	305.4	2.73	6,529	56.30	17,194
LOX(NBP)/LC_2H_4(NBP)	311.5	2.37	6,763	54.88	17,095
LOX(NBP)/LC_3H_6(NBP)	306.8	2.47	6,699	56.92	17,463
LOX(NBP)/L_1-C_4H_8(77°F)	305.8	2.49	6,683	56.20	17,186
LOX(NBP)/RP-1(77°F)	300.4	2.59	6,602	63.91	19,198
N_2O_4/Aerozine-50	288.3	2.0	6,054	74.68	21,530



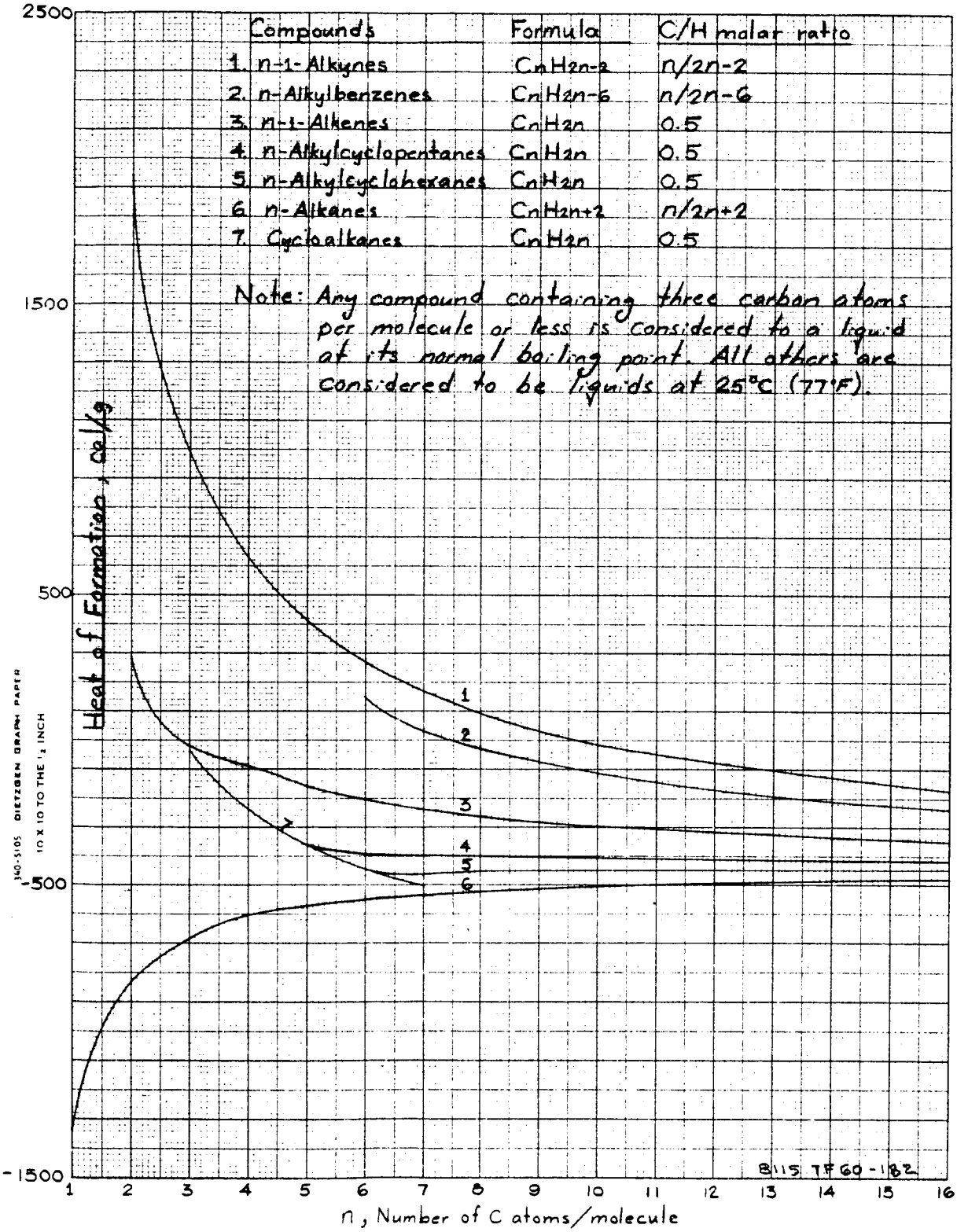
Theoretical Performance and Physical Characteristics of Fuel Rich Gas Generator

Figure II-1



Theoretical Performance of LO_2 (n.b.p.)/Hydrocarbons of Mixture Ratios Which Yield Maximum Specific Impulse

Figure II-2



Heats of Formation of Hydrocarbons

Figure II-3

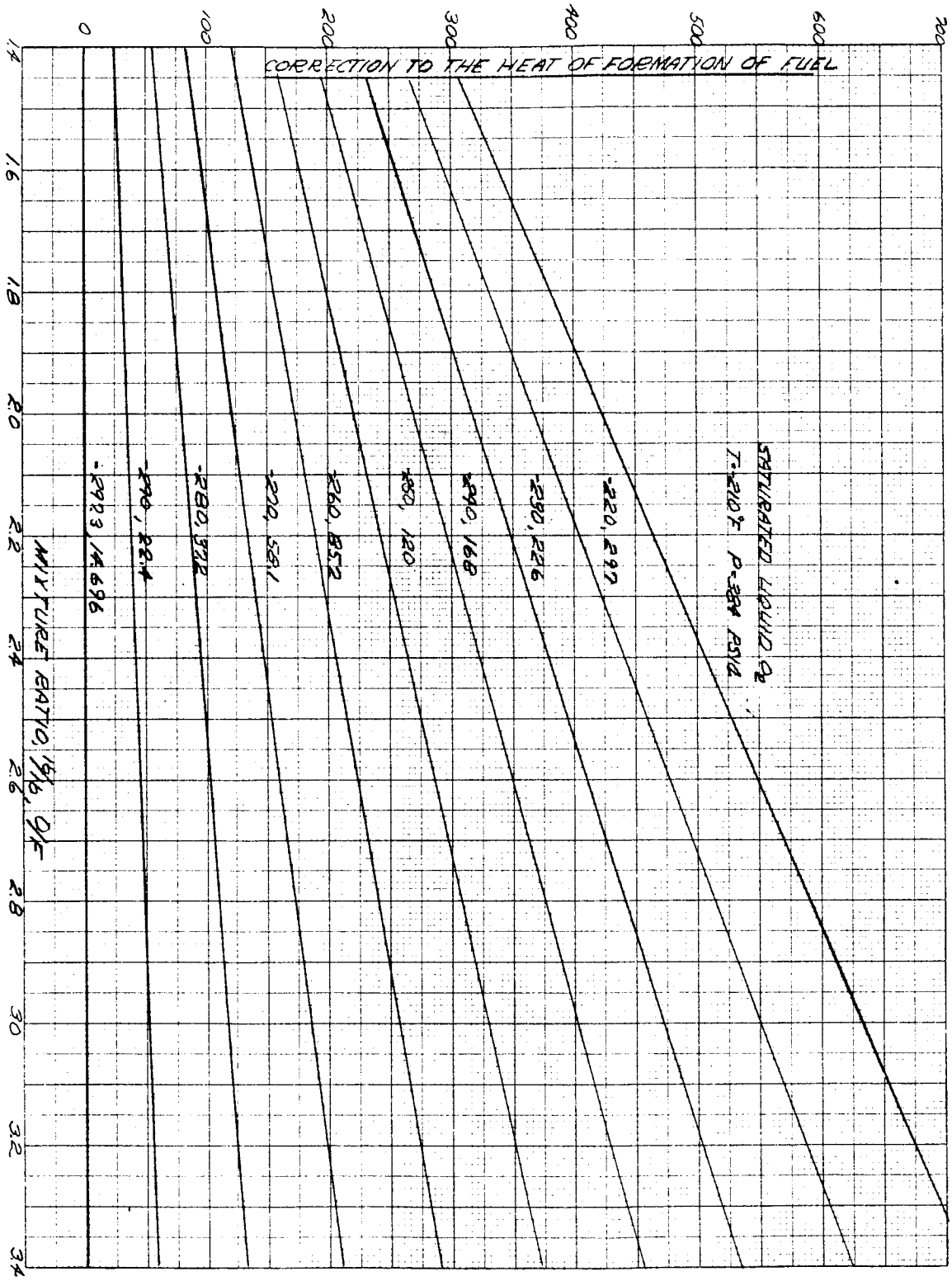
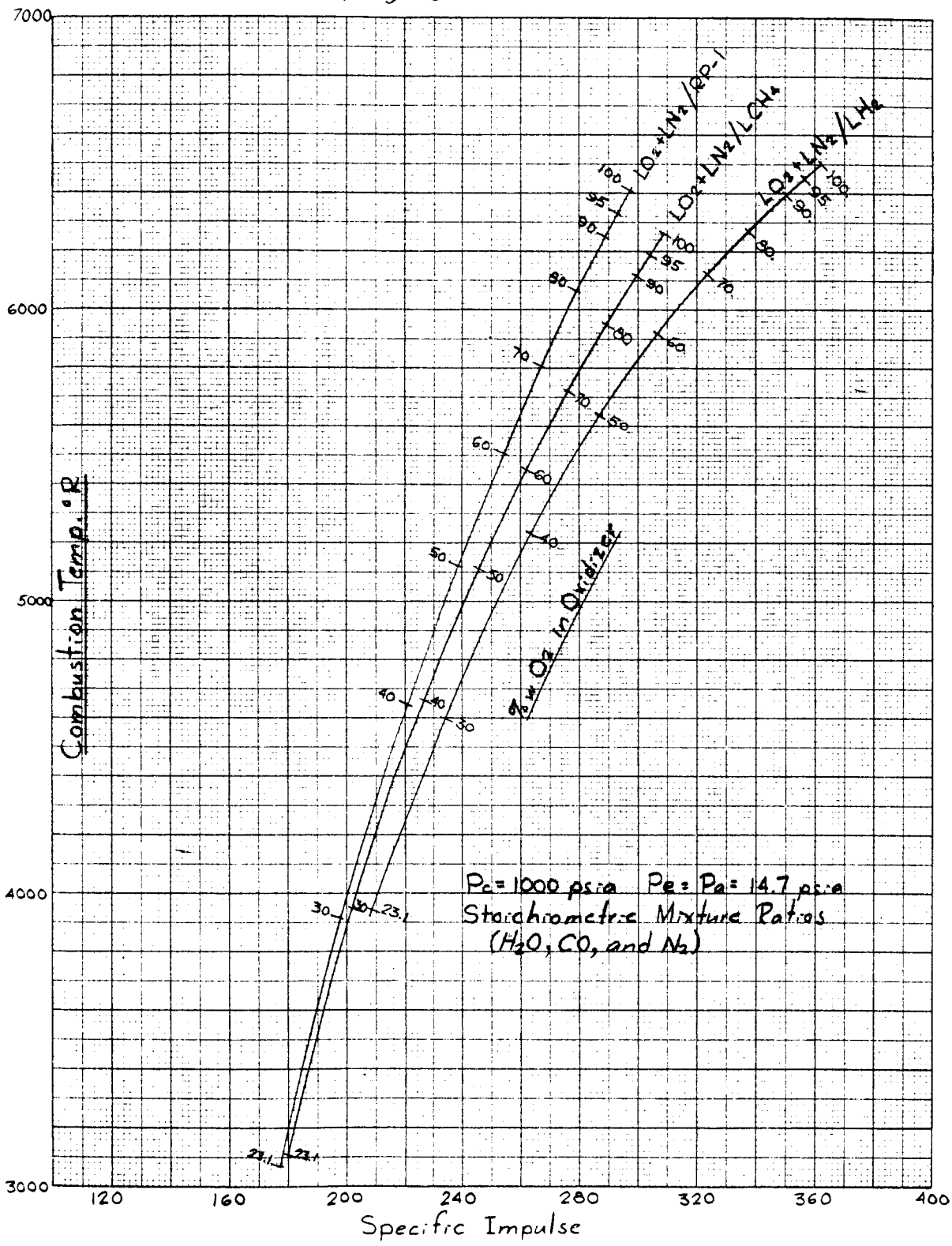

CORRECTION TO THE HEAT OF FORMATION OF FUEL
SATURATED LIQUID O2
T=210°F, P=384 PSIA
-220, 297
-230, 226
-240, 168
-250, 120
-260, 852
-270, 581
-280, 322
-290, 224
-298.3, 146.96
MIXTURE RATIO, lb O/lb F
0
100
200
300
400
500
600
700
1.4
1.6
1.8
2.0
2.2
2.4
2.6
2.8
3.0
3.2
3.4

Figure II-4



Specific Impulses and Combustion Temperatures of LO_2 + LN_2/RP-1, CH_4, and LH_2

Figure II-5

III. ENGINE DESIGN HANDBOOK DATA

A. EXPLANATION OF ENGINE DESIGN HANDBOOK FORMAT

Each page of this handbook has the propellant title listed at the top and is followed by several sets of numbers. The top line of each set lists the chamber pressure (P_c), area ratio ($A_e A_t$), altitude, and the tape identification number. The second line of headings identifies the various performance parameter calculated:

M.R., mixture ratio, the weight ratio of oxidizer to fuel

c*, characteristic velocity, ft/sec

V/c*, ratio of exhaust velocity to characteristic velocity

PA/PCAT, the ratio of exhaust pressure times exhaust area to chamber pressure times thrust area

CF(VAC), the vacuum thrust coefficient

CF(ALT), the thrust coefficient at altitude given in top line

$I_{s(vac)}$, vacuum specific impulse

TC, chamber temperature, °R

T, exhaust temperature, °R

The vacuum specific impulse is curve-fitted as a function of mixture ratio. If a maximum occurs within the given mixture ratio range, the performance parameters for the optimum mixture ratio are printed out on the last line of each set of numbers. If a maximum is not properly defined, such a statement is printed and an optimum is determined based upon a three-point curve-fit of the three points nearest the optimum. Such a value may not be valid. The thrust coefficients and specific impulses listed are 100% theoretical. If nozzle divergence factors and combustion efficiencies are known, they can be applied to the appropriate performance parameters to reflect nonideal performance.

(End text)

TABLE III-1

LOX/LH_2, $P_c = 50$

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

LOX/H2 JAN 5,1960

PC= 50 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	.687	.558	1.245	.951	309.6	236.5	4399	3972
4.00	7899	.667	.570	1.237	.943	303.8	231.6	5091	4744
5.00	7657	.655	.577	1.233	.939	293.5	223.5	5479	5198
6.00	7371	.650	.580	1.230	.936	281.9	214.6	5671	5425
7.00	7083	.647	.583	1.230	.936	270.8	206.1	5746	5514
9.00	6588	.648	.582	1.230	.936	251.8	191.6	5732	5502

PC= 50 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.234	.227	1.461	.873	363.4	217.2	4399	2874
4.00	7899	1.221	.246	1.468	.880	360.4	216.0	5091	3738
5.00	7657	1.200	.267	1.467	.879	349.2	209.3	5479	4431
6.00	7371	1.207	.272	1.479	.892	339.0	204.3	5671	4786
7.00	7083	1.207	.275	1.482	.895	326.4	197.0	5746	4937
9.00	6588	1.204	.277	1.481	.893	303.3	182.9	5732	4936

PC= 50 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.363	.179	1.541	.660	383.4	164.1	4399	2494
4.00	7899	1.357	.198	1.555	.673	381.7	165.2	5091	3331
5.00	7657	1.343	.207	1.551	.669	369.1	159.2	5479	4093
6.00	7371	1.352	.213	1.565	.684	358.7	156.6	5671	4533
7.00	7083	1.353	.217	1.570	.689	345.8	151.6	5746	4728
9.00	6588	1.353	.217	1.571	.689	321.6	141.0	5732	4729

PC= 50 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.435	.152	1.587	.412	394.8	102.4	4399	2258
4.00	7899	1.435	.167	1.602	.426	393.3	104.6	5091	3064
5.00	7657	1.435	.186	1.621	.445	385.9	106.0	5479	3826
6.00	7371	1.433	.201	1.633	.458	374.3	104.9	5671	4358
7.00	7083	1.434	.207	1.641	.465	361.3	102.4	5746	4591
9.00	6588	1.433	.208	1.641	.465	336.0	95.2	5732	4597

PC= 50 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.483	.134	1.617	.147	402.2	36.6	4399	2089
4.00	7899	1.487	.152	1.639	.169	402.4	41.5	5091	2869
5.00	7657	1.490	.172	1.662	.193	395.6	45.8	5479	3638
6.00	7371	1.485	.188	1.673	.204	383.4	46.6	5671	4230
7.00	7083	1.491	.194	1.685	.215	371.0	47.4	5746	4490
9.00	6588	1.491	.195	1.686	.216	345.2	44.2	5732	4499

PC= 50 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.517	.125	1.642	.000	408.5	.0	4399	1962
4.00	7899	1.525	.142	1.667	.000	409.4	.0	5091	2719
5.00	7657	1.533	.157	1.690	.000	402.2	.0	5479	3485
6.00	7371	1.528	.175	1.703	.000	390.2	.0	5671	4116
7.00	7083	1.535	.181	1.716	.000	377.9	.0	5746	4407
9.00	6588	1.535	.182	1.717	.000	351.7	.0	5732	4420

TABLE III-1 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

LOX/H2 JAN 5,1960

PC= 50 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.545	.115	1.659	.000	412.8	.0	4399	1859
4.00	7899	1.555	.131	1.687	.000	414.2	.0	5091	2596
5.00	7657	1.565	.151	1.716	.000	408.4	.0	5479	3356
6.00	7371	1.566	.169	1.735	.000	397.6	.0	5671	4004
7.00	7083	1.568	.178	1.746	.000	384.4	.0	5746	4340
9.00	6588	1.567	.179	1.746	.000	357.6	.0	5732	4358

PC= 50 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.566	.109	1.675	.000	416.8	.0	4399	1775
4.00	7899	1.579	.126	1.705	.000	418.6	.0	5091	2494
5.00	7657	1.592	.142	1.734	.000	412.7	.0	5479	3246
6.00	7371	1.597	.160	1.757	.000	402.5	.0	5671	3911
7.00	7083	1.597	.170	1.767	.000	389.2	.0	5746	4279
9.00	6588	1.596	.171	1.768	.000	362.0	.0	5732	4302

PC= 50 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.600	.099	1.699	.000	422.5	.0	4399	1640
4.00	7899	1.617	.114	1.731	.000	425.1	.0	5091	2329
5.00	7657	1.634	.132	1.765	.000	420.2	.0	5479	3065
6.00	7371	1.641	.150	1.791	.000	410.4	.0	5671	3757
7.00	7083	1.644	.161	1.804	.000	397.3	.0	5746	4177
9.00	6588	1.644	.162	1.806	.000	369.9	.0	5732	4206

PC= 50 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.625	.091	1.716	.000	427.0	.0	4399	1536
4.00	7899	1.645	.105	1.750	.000	429.7	.0	5091	2206
5.00	7657	1.670	.122	1.792	.000	426.5	.0	5479	2903
6.00	7371	1.676	.140	1.816	.000	416.1	.0	5671	3628
7.00	7083	1.680	.152	1.832	.000	403.5	.0	5746	4091
9.00	6588	1.681	.154	1.835	.000	375.7	.0	5732	4128

PC= 50 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.652	.083	1.735	.000	431.6	.0	4399	1419
4.00	7899	1.672	.098	1.770	.000	434.7	.0	5091	2075
5.00	7657	1.702	.114	1.816	.000	432.3	.0	5479	2750
6.00	7371	1.716	.130	1.846	.000	423.1	.0	5671	3465
7.00	7083	1.723	.144	1.866	.000	410.9	.0	5746	3981
9.00	6588	1.723	.146	1.869	.000	382.7	.0	5732	4033

PC= 50 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.684	.073	1.757	.000	437.1	.0	4399	1280
4.00	7899	1.713	.087	1.800	.000	441.9	.0	5091	1877
5.00	7657	1.744	.102	1.846	.000	439.4	.0	5479	2547
6.00	7371	1.763	.119	1.883	.000	431.4	.0	5671	3252
7.00	7083	1.770	.135	1.905	.000	419.4	.0	5746	3841
9.00	6588	1.772	.137	1.909	.000	390.9	.0	5732	3914

TABLE III-1 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

LOX/H2 JAN 5,1960

PC= 50 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.723	.062	1.784	.000	443.9	.0	4399	1104
4.00	7899	1.758	.074	1.832	.000	449.9	.0	5091	1646
5.00	7657	1.796	.089	1.884	.000	448.5	.0	5479	2275
6.00	7371	1.823	.105	1.928	.000	441.8	.0	5671	2958
7.00	7083	1.835	.122	1.957	.000	430.9	.0	5746	3614
9.00	6588	1.837	.125	1.962	.000	401.8	.0	5732	3737

PC= 50 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.746	.055	1.801	.000	448.0	.0	4399	994
4.00	7899	1.786	.066	1.852	.000	454.7	.0	5091	1497
5.00	7657	1.828	.080	1.909	.000	454.4	.0	5479	2095
6.00	7371	1.861	.096	1.957	.000	448.4	.0	5671	2756
7.00	7083	1.877	.113	1.990	.000	438.2	.0	5746	3440
9.00	6588	1.879	.117	1.996	.000	408.8	.0	5732	3607

PC= 50 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8002	1.763	.050	1.813	.000	450.9	.0	4399	915
4.00	7899	1.806	.061	1.866	.000	458.3	.0	5091	1388
5.00	7657	1.852	.074	1.927	.000	458.6	.0	5479	1959
6.00	7371	1.888	.089	1.977	.000	453.1	.0	5671	2606
7.00	7083	1.908	.106	2.014	.000	443.5	.0	5746	3296
9.00	6588	1.910	.112	2.022	.000	414.0	.0	5732	3500

TABLE III-2

LOX/LH_2, $P_c = 100$

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 100 AE/AT= 1ATMOS ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.172	.250	1.421	1.172	353.8	291.6	4432	3043
4.00	7935	1.173	.260	1.432	1.173	353.3	289.2	5183	3880
5.00	7711	1.175	.269	1.444	1.175	346.5	281.9	5614	4531
6.00	7439	1.177	.275	1.452	1.177	335.9	272.2	5831	4925
7.00	7154	1.178	.278	1.456	1.178	323.7	262.0	5916	5090
9.00	6644	1.178	.278	1.456	1.178	300.8	243.3	5894	5079

PC= 100 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	.690	.556	1.246	1.099	310.1	273.6	4432	3984
4.00	7935	.670	.568	1.238	1.091	305.5	269.2	5183	4807
5.00	7717	.658	.575	1.233	1.086	295.9	260.6	5614	5306
6.00	7439	.652	.579	1.231	1.084	284.7	250.7	5831	5564
7.00	7154	.651	.579	1.230	1.083	273.6	240.9	5916	5665
9.00	6644	.650	.580	1.230	1.083	254.1	223.7	5894	5647

PC= 100 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.364	.178	1.541	1.101	383.7	273.9	4432	2486
4.00	7935	1.358	.195	1.552	1.112	383.0	274.2	5183	3317
5.00	7717	1.351	.214	1.564	1.123	375.3	269.5	5614	4076
6.00	7439	1.348	.212	1.560	1.120	360.8	258.9	5831	4601
7.00	7154	1.346	.218	1.563	1.123	347.7	249.7	5916	4831
9.00	6644	1.345	.219	1.564	1.123	323.0	232.0	5894	4828

PC= 100 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.436	.152	1.587	.999	395.0	248.7	4432	2250
4.00	7935	1.435	.164	1.599	1.011	394.5	249.5	5183	3045
5.00	7711	1.433	.183	1.616	1.028	387.7	246.7	5614	3816
6.00	7439	1.430	.199	1.629	1.041	376.7	240.7	5831	4404
7.00	7154	1.428	.206	1.634	1.046	363.4	232.7	5916	4683
9.00	6644	1.428	.207	1.635	1.047	337.7	216.3	5894	4685

PC= 100 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.484	.133	1.617	.882	402.4	219.5	4432	2082
4.00	7935	1.486	.149	1.635	.901	403.4	222.2	5183	2849
5.00	7717	1.489	.168	1.657	.922	397.4	221.1	5614	3617
6.00	7439	1.488	.185	1.672	.937	386.7	216.8	5831	4249
7.00	7154	1.486	.193	1.679	.944	373.4	210.0	5916	4571
9.00	6644	1.486	.194	1.680	.945	347.0	195.2	5894	4578

PC= 100 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.518	.124	1.642	.760	408.7	189.3	4432	1954
4.00	7935	1.524	.140	1.664	.782	410.4	192.9	5183	2698
5.00	7717	1.531	.153	1.684	.802	404.0	192.4	5614	3456
6.00	7439	1.532	.170	1.702	.821	393.7	189.8	5831	4118
7.00	7154	1.531	.180	1.711	.829	380.5	184.4	5916	4479
9.00	6644	1.531	.182	1.712	.830	353.6	171.5	5894	4492

TABLE III-2 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 100 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.545	.114	1.659	.631	413.0	156.9	4432	1851
4.00	7935	1.554	.129	1.683	.654	415.1	161.3	5183	2575
5.00	7717	1.562	.147	1.710	.681	410.2	163.4	5614	3322
6.00	7439	1.566	.165	1.731	.702	400.3	162.4	5831	4006
7.00	7154	1.565	.176	1.741	.712	387.2	158.4	5916	4404
9.00	6644	1.565	.178	1.743	.714	359.9	147.4	5894	4422

PC= 100 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.567	.108	1.675	.500	417.0	124.4	4432	1766
4.00	7935	1.577	.123	1.701	.525	419.5	129.5	5183	2474
5.00	7717	1.589	.138	1.727	.552	414.4	132.4	5614	3209
6.00	7439	1.595	.156	1.751	.575	404.9	133.0	5831	3906
7.00	7154	11594	.168	1.763	.587	392.0	130.5	5916	4337
9.00	6644	1.594	.170	1.764	.589	364.4	121.6	5894	4361

PC= 100 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.600	.098	1.698	.229	422.7	56.9	4432	1632
4.00	7935	1.614	.112	1.727	.257	425.9	63.4	5183	2311
5.00	7717	1.630	.128	1.758	.289	421.8	69.2	5614	3026
6.00	7439	1.639	.146	1.785	.316	412.9	73.0	5831	3736
7.00	7154	1.640	.159	1.799	.330	400.1	73.3	5916	4224
9.00	6644	1.640	.161	1.801	.332	372.0	68.5	5894	4260

PC= 100 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.625	.091	1.716	.000	427.0	.0	4432	1530
4.00	7935	1.642	.104	1.746	.000	430.6	.0	5183	2183
5.00	7717	1.661	.120	1.781	.017	427.2	4.2	5614	2883
6.00	7439	1.674	.138	1.811	.048	418.8	11.0	5831	3595
7.00	7154	1.676	.150	1.826	.063	406.1	13.9	5916	4128
9.00	6644	1.676	.152	1.829	.065	377.7	13.5	5894	4177

PC= 100 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.653	.082	1.735	.000	431.8	.0	4432	1412
4.00	7935	1.673	.095	1.768	.000	436.2	.0	5183	2037
5.00	7711	1 696	.111	1.807	.000	433.5	.0	5614	2714
6.00	7439	1.713	.126	1.839	.000	425.4	.0	5831	3423
7.00	7154	1.717	.142	1.859	.000	413.4	.0	5916	4008
9.00	6644	1.717	.145	1.862	.000	384.6	.0	5894	4075

PC= 100 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.684	.073	1.757	.000	437.3	.0	4432	1273
4.00	7935	1.709	.085	1.794	.000	442.6	.0	5183	1859
5.00	7717	1.737	.100	1.837	.000	440.6	.0	5614	2507
6.00	7439	1.759	.115	1.875	.000	433.5	.0	5831	3202
7.00	7154	1.767	.132	1.898	.000	422.2	.0	5916	3842
9.00	6644	1.767	.136	1.903	.000	392.9	.0	5894	3943

TABLE III-2 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 100 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3800	8003	13723	.061	1.754	.000	444.1	.0	4432	1097
4.00	7935	1.753	.073	1.826	.000	450.5	.0	5183	1629
5.00	7717	1.788	.086	1.874	.000	449.6	.0	5614	2237
6.00	7439	1.817	.102	1.919	.000	443.8	.0	5831	2902
7.00	7154	1.831	.118	1.948	.000	433.5	.0	5916	3585
9.00	6644	1.831	.124	1.955	.000	403.7	.0	5894	3749

PC= 100 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.746	.054	1.801	.000	448.2	.0	4432	987
4.00	7935	1.781	.065	1.846	.000	455.3	.0	5183	1481
5.00	7717	1.820	.078	1.898	.000	455.3	.0	5614	2059
6.00	7439	1.854	.093	1.947	.000	450.2	.0	5831	2700
7.00	7154	1.873	.109	1.982	.000	440.8	.0	5916	3391
9.00	6644	1.874	.115	1.989	.000	410.8	.0	5894	3604

PC= 100 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.763	.050	1.812	.000	451.1	.0	4432	909
4.00	7935	1.800	.060	1.863	.000	458.8	.0	5183	1374
5.00	7717	1.843	.072	1.915	.000	459.4	.0	5614	1928
6.00	7439	1.881	.086	1.967	.000	454.8	.0	5831	2550
7.00	7154	12903	.103	2.006	.000	446.0	.0	5916	3237
9.00	6644	1.905	.110	2.015	.000	416.1	.0	5894	3487

TABLE III-3

LOX/LH_2, P_c = 300

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIMB)DD APR.16,1959

PC= 300 AE/AT= 1ATMOS ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.395	.165	1.560	1.395	388.1	347.1	4448	2390
4.00	7953	1.405	.174	1.580	1.405	390.5	347.4	5272	3141
5.00	7763	1.418	.185	1.603	1.418	387.1	342.5	5786	3843
6.00	7510	1.430	.195	1.626	1.430	379.5	333.9	6064	4431
7.00	7237	1.436	.201	1.637	1.436	368.4	323.1	6179	4775
9.00	6729	1.437	.202	1.639	1.437	342.9	300.6	6160	4788

PC= 300 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	.692	.555	1.247	1.198	310.2	298.0	4448	3985
4.00	7953	.674	.565	1.240	1.191	306.5	294.4	5272	4654
5.00	7767	.661	.574	1.235	1.186	298.2	286.3	5786	5436
6.00	7510	.655	.577	1.233	1.184	287.8	276.4	6064	5761
7.00	7237	.652	.580	1.231	1.182	277.0	266.0	6179	5899
9.00	6729	.653	.578	1.231	1.182	257.5	247.3	6160	5882

PC= 300 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.365	.178	1.542	1.395	383.7	347.1	4448	2488
4.00	7953	1.358	.193	1.551	1.404	383.5	347.2	5272	3303
5.00	7767	1.352	.209	1.562	1.415	377.1	341.6	5786	4062
6.00	7510	1.350	.209	1.559	1.412	363.9	329.6	6064	4654
7.00	7237	1.347	.215	1.562	1.415	351.5	318.4	6179	4959
9.00	6729	1.347	.216	1.563	1.416	327.0	296.2	6160	4966

PC= 300 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	12436	.152	1.568	1.392	395.0	346.3	4448	2253
4.00	7953	1.435	.162	1.597	1.402	395.0	346.5	5272	3033
5.00	7767	1.434	.179	1.612	1.417	389.3	342.0	5786	3789
6.00	7510	1.432	.194	1.626	1.430	379.6	333.9	6064	4427
7.00	7237	1.429	.204	1.632	1.436	367.2	323.1	6179	4791
9.00	6729	1.428	.205	1.633	1.437	341.6	300.7	6160	4806

PC= 300 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.484	.133	1.618	1.373	402.4	341.5	4448	2084
4.00	7953	1.486	.148	1.633	1.388	403.8	343.3	5272	2839
5.00	7767	1.488	.164	1.652	1.407	398.9	339.8	5786	3584
6.00	7510	1.488	.180	1.669	1.424	389.6	332.4	6064	4250
7.00	7237	1.486	.190	1.677	1.432	377.2	322.1	6179	4662
9.00	6729	1.486	.192	1.678	1.433	351.0	299.8	6160	4686

PC= 300 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.519	.124	1.643	1.349	408.7	335.6	4448	1957
4.00	7953	1.522	.139	1.661	1.367	410.8	338.1	5272	2690
5.00	7767	1.528	.155	1.682	1.389	406.2	335.3	5786	3423
6.00	7510	1.532	.166	1.698	1.404	396.4	327.8	6064	4103
7.00	7237	1.531	.177	1.708	1.414	384.2	318.1	6179	4556
9.00	6729	1.531	.179	1.710	1.416	357.6	296.1	6160	4589

TABLE III-3 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LHQUI4) APR.16,1959

PC= 300 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	12546	.114	1.660	1.317	413.0	327.7	4448	1853
4.00	7953	1.552	.128	1.680	1.337	415.5	330.7	5272	2567
5.00	7767	1.560	.144	1.704	1.361	411.5	328.7	5786	3289
6.00	7510	1.565	.161	1.726	1.383	402.9	322.9	6064	3979
7.00	7237	1.565	.173	1.738	1.395	390.9	313.8	6179	4468
9.00	6729	1.565	.175	1.740	1.397	363.9	292.2	6160	4510

PC= 300 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.568	.109	1.676	1.284	417.0	319.5	4448	1768
4.00	7953	1.576	.123	1.698	1.306	419.9	323.0	5272	2467
5.00	7767	1.586	.135	1.721	1.329	415.6	321.0	5786	3176
6.00	7510	1.594	.151	1.745	1.352	407.4	315.9	6064	3871
7.00	7237	1.594	.165	1.759	1.367	395.7	307.5	6179	4389
9.00	6729	1.594	.167	1.761	1.369	368.4	286.4	6160	4440

PC= 300 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.601	.098	1.699	1.209	422.7	300.9	4448	1633
4.00	7953	1.612	.112	1.724	1.234	426.3	305.1	5272	2305
5.00	7767	1.626	.125	1.752	1.262	422.9	304.6	5786	2995
6.00	7510	1.637	.141	1.779	1.289	415.2	300.9	6064	3690
7.00	7233	1.639	.155	1.795	1.305	403.8	293.6	6179	4255
9.00	6729	1.639	.158	1.797	1.307	376.0	272.5	6160	4225

PC= 300 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.626	.091	1.717	1.129	427.1	280.9	4448	1531
4.00	7953	1.640	.103	1.743	1.155	430.9	285.6	5272	2178
5.00	7767	1.657	.117	1.774	1.186	428.3	286.4	5786	2854
6.00	7510	1.671	.133	1.804	1.216	421.1	283.8	6064	3544
7.00	7237	1.675	.146	1.821	1.233	409.7	277.4	6179	4142
9.00	6729	1.675	.149	1.824	1.236	381.6	258.6	6160	4230

PC= 300 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.654	.082	1.736	1.001	431.8	249.1	4448	1412
4.00	7953	1.671	.095	1.768	1.031	436.5	254.8	5272	2032
5.00	7767	1.691	.109	1.799	1.065	434.5	257.1	5786	2687
6.00	7510	1.709	.122	1.831	1.096	427.5	255.9	6064	3367
7.00	7237	1.715	.137	1.853	1.118	416.8	251.5	6179	3999
9.00	6729	1.715	.141	1.857	1.122	388.4	234.7	6160	4113

PC= 300 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.685	.073	1.768	.778	437.4	193.6	4448	1273
4.00	7953	1.707	.085	1.791	.812	442.9	200.7	5272	1854
5.00	7767	1.731	.098	1.829	.849	441.5	205.0	5786	2483
6.00	7510	1.753	.112	1.866	.886	435.6	206.9	6064	3146
7.00	7237	1.764	.127	1.881	.911	425.5	205.1	6179	3805
9.00	6729	1.764	.132	1.896	.917	396.7	191.7	6160	3960

TABLE III-3 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 300 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.724	.061	1.785	.316	444.1	78.5	4448	1097
4.00	7953	1.751	.072	1.823	.353	450.7	87.4	5272	1624
5.00	7767	1.781	.085	1.866	.396	450.5	95.7	5786	2216
6.00	7510	1.810	.098	1.908	.439	445.5	102.4	6064	2850
7.00	7237	1.827	.113	1.940	.471	436.5	105.9	6179	3519
9.00	6729	1.828	.120	1.947	.478	407.3	99.9	6160	3736

PC= 300 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.747	.054	1.802	.000	448.2	.0	4448	986
4.00	7953	1.778	.065	1.843	.000	455.7	.0	5272	1477
5.00	7767	1.812	.077	1.889	.000	456.0	.0	5786	2039
6.00	7510	1.845	.090	1.935	.000	451.8	.0	6064	2652
7.00	7237	1.867	.105	1.972	.012	443.6	2.7	6179	3314
9.00	6729	1.869	.112	1.981	.021	414.3	4.5	6160	3570

PC= 300 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8003	1.764	.050	1.813	.000	451.1	.0	4448	908
4.00	7953	1.797	.059	1.856	.000	459.0	.0	5272	1369
5.00	7767	1.834	.071	1.906	.000	460.1	.0	5786	1909
6.00	7510	1.871	.083	1.954	.000	456.2	.0	6064	2505
7.00	7237	1.896	.098	1.994	.000	448.6	.0	6179	3157
9.00	6729	1.899	.106	2.005	.000	419.4	.0	6160	3436

TABLE III-4

LOX/LH_2, P_c = 500

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 500 AE/AT= 1ATMOS ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.472	.139	1.610	1.472	401.0	366.5	4469	2122
4.00	7973	1.486	.147	1.633	1.486	404.8	368.3	5333	2823
5.00	7809	1.503	.157	1.660	1.503	402.9	364.7	5886	3503
6.00	7566	1.520	.168	1.688	1.520	397.0	357.4	6189	4135
7.00	7292	1.531	.177	1.708	1.531	387.1	347.0	6315	4591
9.00	6767	1.533	.179	1.712	1.533	360.1	322.5	6284	4622

PC= 500 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	.692	.555	1.247	1.218	310.5	303.2	4469	3995
4.00	7973	.677	.564	1.241	1.212	307.6	300.3	5333	4891
5.00	7809	.663	.572	1.236	1.206	299.9	292.8	5886	5512
6.00	7566	.656	.577	1.233	1.203	289.9	283.0	6189	5868
7.00	7292	.653	.579	1.232	1.202	279.2	272.6	6315	6019
9.00	6767	.652	.580	1.232	1.202	259.1	252.9	6284	5994

PC= 500 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.365	.177	1.542	1.453	383.9	361.9	4469	2482
4.00	7973	1.359	.191	1.550	1.462	384.2	362.4	5333	3292
5.00	7809	1.353	.207	1.560	1.472	378.6	357.2	5886	4058
6.00	7566	1.351	.206	1.557	1.469	366.2	345.4	6189	4683
7.00	7292	1.348	.214	1.562	1.473	354.0	334.0	6315	5025
9.00	6767	1.347	.215	1.562	1.474	328.7	310.1	6284	5028

PC= 500 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8012	18436	.151	1.587	1.470	395.2	365.9	4469	2247
4.00	7973	1.435	.161	1.596	1.479	395.6	366.5	5333	3019
5.00	7809	1.434	.176	1.610	1.492	390.7	362.2	5886	3775
6.00	7566	1.432	.191	1.623	1.505	381.7	354.1	6189	4439
7.00	7292	1.429	.202	1.631	1.513	369.7	343.0	6315	4845
9.00	6767	1.429	.204	1.632	1.515	343.4	318.7	6284	4858

PC= 500 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.484	.133	1.617	1.470	402.6	366.0	4469	2079
4.00	7973	1.486	.146	1.632	1.485	404.5	368.0	5333	2824
5.00	7809	1.487	.161	1.649	1.502	400.2	364.5	5886	3564
6.00	7566	1.488	.177	1.665	1.518	391.6	357.1	6189	4249
7.00	7292	1.486	.189	1.675	1.528	379.7	346.4	6315	4707
9.00	6767	1.486	.191	1.677	1.530	352.8	321.8	6284	4731

PC= 500 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.518	.124	1.642	1.466	408.9	365.0	4469	1952
4.00	7973	1.522	.137	1.660	1.483	411.3	367.6	5333	2675
5.00	7809	1.527	.152	1.679	1.502	407.5	364.7	5886	3400
6.00	7566	1.531	.163	1.694	1.518	398.4	356.9	6189	4094
7.00	7292	1.531	.175	1.706	1.530	386.7	346.7	6315	4592
9.00	6767	1.531	.177	1.708	1.532	359.3	322.2	6284	4629

TABLE III-4 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 500 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.546	.114	1.659	1.453	413.1	361.9	4469	1848
4.00	7973	1.552	.127	1.678	1.473	416.0	365.0	5333	2552
5.00	7809	1.559	.141	1.700	1.494	412.6	362.7	5886	3265
6.00	7566	1.564	.157	1.722	1.516	404.9	356.5	6189	3963
7.00	7292	1.565	.171	1.735	1.530	393.4	346.8	6315	4497
9.00	6767	1.564	.174	1.738	1.532	365.6	322.3	6284	4545

PC= 500 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.567	.108	1.675	1.440	417.1	358.6	4469	1763
4.00	7973	1.575	.121	1.696	1.461	420.4	362.1	5333	2452
5.00	7809	1.584	.132	1.717	1.482	416.8	359.7	5886	3151
6.00	7566	1.592	.148	1.740	1.505	409.3	354.0	6189	3849
7.00	7292	1.594	.163	1.756	1.521	398.1	344.8	6315	4411
9.00	6767	1.594	.166	1.759	1.524	370.1	320.6	6284	4471

PC= 500 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.600	.098	1.698	1.404	422.8	349.6	4469	1629
4.00	7973	1.611	.111	1.722	1.428	426.7	353.9	5333	2290
5.00	7809	1.624	.123	1.747	1.453	424.0	352.6	5886	2969
6.00	7566	1.635	.138	1.773	1.479	417.0	347.9	6189	3662
7.00	7292	1.639	.153	1.792	1.498	406.2	339.5	6315	4265
9.00	6767	1.639	.156	1.795	1.501	377.7	315.8	6284	4348

PC= 500 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.625	.091	1.715	1.363	427.2	339.3	4469	1527
4.00	7973	1.638	.102	1.740	1.387	431.3	343.9	5333	2164
5.00	7809	1.654	.115	1.769	1.416	429.3	343.7	5886	2826
6.00	7566	1.668	.130	1.798	1.445	422.8	339.8	6189	3511
7.00	7292	1.674	.144	1.818	1.465	412.0	332.1	6315	4142
9.00	6767	1.674	.148	1.822	1.469	383.3	309.1	6284	4247

PC= 500 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.652	.082	1.734	1.294	431.9	322.1	4469	1410
4.00	7973	1.669	.094	1.762	1.322	436.8	327.6	5333	2018
5.00	7809	1.687	.107	1.794	1.353	435.4	328.4	5886	2660
6.00	7566	1.705	.121	1.826	1.385	429.4	325.7	6189	3331
7.00	7292	1.714	.135	1.849	1.408	419.1	319.1	6315	3986
9.00	6767	1.715	.140	1.854	1.413	390.1	297.3	6284	4123

PC= 500 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.684	.073	1.757	1.169	437.4	291.0	4469	1271
4.00	7973	1.704	.084	1.788	1.200	443.2	297.5	5333	1841
5.00	7809	1.727	.096	1.822	1.235	442.4	299.7	5886	2456
6.00	7566	1.749	.110	1.858	1.271	437.1	298.8	6189	3107
7.00	7292	1.762	.124	1.886	1.299	427.6	294.4	6315	3777
9.00	6767	1.763	.130	1.893	1.305	398.3	274.6	6284	3960

TABLE III-4 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 2HD 05 JUNE 1961

LOX/H2(LIQUID) APR.16,1959

PC= 500 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.722	.061	1.784	.902	444.1	224.6	4469	1096
4.00	7973	1.748	.072	1.820	.938	451.0	232.4	5333	1614
5.00	7809	1.775	.083	1.859	.977	451.2	237.2	5886	2190
6.00	7566	1.804	.096	1.900	1.018	446.8	239.4	6189	2810
7.00	7292	1.824	.110	1.934	1.052	438.5	238.6	6315	3476
9.00	6767	1.826	.118	1.944	1.062	408.9	223.4	6284	3722

PC= 500 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.746	.054	1.800	.624	448.3	155.5	4469	985
4.00	7973	1.775	.065	1.839	.664	455.9	164.5	5333	1467
5.00	7809	1.806	.075	1.881	.706	456.7	171.3	5886	2015
6.00	7566	1.838	.088	1.927	.751	453.1	176.6	6189	2613
7.00	7292	1.863	.102	1.965	.789	445.4	178.9	6315	3265
9.00	6767	1.867	.110	1.976	.801	415.8	168.5	6284	3547

PC= 500 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8010	1.762	.050	1.812	.342	451.1	85.2	4469	907
4.00	7973	1.794	.059	1.853	.383	459.2	94.9	5333	1361
5.00	7809	1.828	.070	1.898	.428	460.6	103.9	5886	1886
6.00	7566	1.866	.075	1.941	.472	456.5	110.9	6189	2454
7.00	7292	1.892	.095	1.987	.517	450.3	117.2	6315	3105
9.00	6767	1.897	.104	2.000	.531	420.8	111.6	6284	3408

TABLE III-5

LOX/LH_2, P_c = 700

ENGINE DESIGN HANDBOOK DATA. TAPE 12 05 JUNE 1961

LOX/H2 JAN 5,1960

PC= 700 AE/AT= 1ATMOS ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.517	.125	1.641	1.517	408.3	377.3	4458	1962
4.00	7968	1.533	.133	1.666	1.533	412.7	379.6	5334	2633
4.50	7897	1.544	.138	1.682	1.544	412.9	379.1	5660	2957
5.00	7804	1.553	.143	1.696	1.553	411.5	376.8	5916	3288
5.50	7695	1.564	.148	1.711	1.564	409.4	374.0	6110	3608
6.00	7571	1.574	.153	1.727	1.574	406.4	370.4	6247	3912
6.50	7441	1.583	.158	1.741	1.583	402.8	366.2	6338	4195

PC= 700 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	.691	.556	1.247	1.226	310.2	305.0	4458	3987
4.00	7968	.676	.564	1.241	1.220	307.3	302.1	5334	4886
4.50	7897	.667	.572	1.238	1.217	304.0	298.9	5660	5245
5.00	7804	.665	.572	1.237	1.216	300.0	294.9	5916	5529
5.50	7695	.659	.576	1.235	1.214	295.4	290.3	6110	5753
6.00	7571	.658	.575	1.233	1.212	290.3	285.4	6247	5911
6.50	7441	.654	.579	1.233	1.212	285.1	280.3	6338	6023

PC= 700 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.365	.178	1.542	1.479	383.6	367.9	4458	2486
4.00	7968	1.358	.192	1.550	1.487	383.9	368.3	5334	3292
4.50	7897	1.356	.199	1.555	1.492	381.8	366.3	5660	3679
5.00	7804	1.354	.206	1.561	1.498	378.6	363.3	5916	4049
5.50	7695	1.352	.213	1.565	1.502	374.5	359.4	6110	4387
6.00	7571	1.352	.205	1.557	1.494	366.5	351.7	6247	4680
6.50	7441	1.345	.212	1.557	1.494	360.2	345.7	6338	4916

PC= 700 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.436	.152	1.588	1.504	395.0	374.1	4458	2251
4.00	7968	1.434	.161	1.596	1.512	395.3	374.5	5334	3022
4.50	7893	1.435	.168	1.603	1.519	393.5	372.9	5660	3400
5.00	7804	1.435	.176	1.610	1.526	390.6	370.2	5916	3768
5.50	7695	1.434	.183	1.617	1.533	386.9	366.8	6110	4117
6.00	7571	1.433	.190	1.623	1.539	382.0	362.3	6247	4429
6.50	7441	1.432	.197	1.628	1.544	376.6	357.2	6338	4689

PC= 700 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.484	.133	1.618	1.513	402.3	376.2	4458	2082
4.00	7968	1.485	.147	1.631	1.526	404.1	378.1	5334	2828
4.50	7897	1.486	.154	1.640	1.535	402.7	376.9	5660	3197
5.00	7804	1.488	.161	1.649	1.544	400.1	374.6	5916	3560
5.50	7695	1.489	.169	1.658	1.553	396.5	371.4	6110	3912
6.00	7571	1.489	.176	1.665	1.560	391.9	367.2	6247	4237
6.50	7441	1.489	.183	1.671	1.566	386.6	362.3	6338	4518

TABLE III-5 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 05 JUNE 1961

LOX/H2 JAN 5,1960

PC= 700 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.518	.124	1.643	1.517	408.6	377.3	4458	1956
4.00	7968	1.521	.138	1.659	1.533	411.0	379.8	5334	2680
4.50	7897	1.524	.145	1.669	1.543	409.8	378.8	5660	3041
5.00	7804	1.527	.152	1.679	1.553	407.3	376.8	5916	3399
5.50	7695	1.531	.154	1.685	1.559	403.0	372.9	6110	3748
6.00	7571	1.532	.162	1.694	1.568	398.6	368.9	6247	4080
6.50	7441	1.532	.169	1.701	1.575	393.5	364.3	6338	4377

PC= 700 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.546	.114	1.660	1.513	412.9	376.3	4458	1852
4.00	7968	1.551	.127	1.678	1.531	415.7	379.3	5334	2557
4.50	7897	1.555	.134	1.689	1.542	414.7	378.6	5660	2912
5.00	7804	1.559	.141	1.700	1.553	412.5	376.8	5916	3264
5.50	7695	1.562	.149	1.711	1.564	409.3	374.1	6110	3614
6.00	7571	1.564	.157	1.721	1.574	405.1	370.5	6247	3949
6.50	7441	1.566	.164	1.730	1.583	400.1	366.1	6338	4258

PC= 700 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.567	.109	1.676	1.508	416.9	375.1	4458	1768
4.00	7968	1.574	.122	1.696	1.528	420.1	378.5	5334	2457
4.50	7897	1.579	.129	1.708	1.540	419.2	377.9	5660	2805
5.00	7804	1.585	.133	1.717	1.549	416.6	375.8	5916	3152
5.50	7695	1.589	.140	1.729	1.561	413.5	373.3	6110	3498
6.00	7571	1.592	.147	1.740	1.572	409.4	369.9	6247	3836
6.50	7441	1.594	.155	1.749	1.581	404.6	365.8	6338	4153

PC= 700 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.601	.098	1.699	1.489	422.6	370.3	4458	1634
4.00	7968	1.610	.111	1.722	1.512	426.4	374.4	5334	2295
4.50	7897	1.617	.117	1.734	1.524	425.6	374.1	5660	2632
5.00	7804	1.625	.123	1.748	1.538	424.0	373.0	5916	2968
5.50	7695	1.630	.130	1.760	1.550	421.0	370.8	6110	3312
6.00	7571	1.635	.138	1.772	1.562	417.1	367.7	6247	3650
6.50	7441	1.638	.145	1.783	1.573	412.4	363.9	6338	3978

PC= 700 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.626	.091	1.717	1.465	427.0	364.3	4458	1531
4.00	7968	1.638	.102	1.740	1.488	431.1	368.7	5334	2169
4.50	7891	1.646	.109	1.755	1.503	430.7	368.9	5660	2498
5.00	7804	1.658	.114	1.772	1.520	429.9	368.8	5916	2808
5.50	7695	1.663	.122	1.785	1.533	426.9	366.6	6110	3155
6.00	7571	1.668	.129	1.797	1.545	423.0	363.7	6247	3495
6.50	7441	1.673	.137	1.810	1.558	418.6	360.3	6338	3826

TABLE III-5 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 05 JUNE 1961

LOX/H2 JAN 5,1960

PC= 700 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.653	.083	1.735	1.421	431.7	353.3	4458	1414
4.00	7968	1.669	.094	1.763	1.448	436.6	358.6	5334	2022
4.50	7897	1.678	.100	1.778	1.463	436.5	359.2	5660	2342
5.00	7804	1.688	.107	1.794	1.479	435.3	358.9	5916	2663
5.50	7695	1.696	.114	1.810	1.495	432.9	357.6	6110	2991
6.00	7571	1.704	.121	1.824	1.509	429.3	355.2	6247	3325
6.50	7441	1.713	.126	1.838	1.523	425.2	352.4	6338	3647

PC= 700 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.685	.073	1.758	1.338	437.2	332.7	4458	1275
4.00	7968	1.704	.084	1.788	1.368	443.0	338.9	5334	1848
4.50	7897	1.715	.090	1.805	1.386	443.2	340.1	5660	2151
5.00	7804	1.727	.096	1.823	1.403	442.2	340.4	5916	2462
5.50	7695	1.738	.103	1.840	1.420	440.2	339.7	6110	2780
6.00	7571	1.751	.108	1.860	1.440	437.7	338.9	6247	3082
6.50	7441	1.757	.115	1.873	1.453	433.2	336.1	6338	3429

PC= 700 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.723	.062	1.784	1.155	443.9	287.2	4458	1100
4.00	7968	1.748	.072	1.820	1.190	450.7	294.7	5334	1620
4.50	7897	1.762	.078	1.840	1.210	451.6	297.0	5660	1903
5.00	7804	1.776	.084	1.859	1.230	451.1	298.3	5916	2197
5.50	7695	1.789	.089	1.879	1.249	449.4	298.7	6110	2499
6.00	7571	11802	.096	1.898	1.268	446.7	298.5	6247	2809
6.50	7441	1.814	.103	1.917	1.287	443.3	297.7	6338	3131

PC= 700 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.746	.055	1.801	.961	448.0	239.1	4458	990
4.00	7968	1.775	.065	1.840	1.000	455.7	247.7	5334	1472
4.50	7897	1.791	.070	1.860	1.021	456.7	250.5	5660	1740
5.00	7804	1.807	.075	1.882	1.042	456.6	252.8	5916	2021
5.50	7695	1.822	.081	1.903	1.064	455.3	254.4	6110	2313
6.00	7571	1.837	.088	1.925	1.085	453.0	255.3	6247	2613
6.50	7441	1.851	.094	1.945	1.105	449.9	255.6	6338	2926

PC= 700 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8001	1.763	.050	1.812	.763	450.8	189.7	4458	912
4.00	7968	1.794	.059	1.853	.803	459.0	198.9	5334	1365
4.50	7897	1.811	.064	1.876	.826	460.4	202.8	5660	1621
5.00	7804	1.830	.070	1.899	.850	460.8	206.1	5916	1884
5.50	7695	1.844	.076	1.920	.870	459.3	208.2	6110	2184
6.00	7571	1.862	.081	1.943	.893	457.3	210.2	6247	2467
6.50	7441	1.877	.087	1.965	.915	454.5	211.7	6338	2774

TABLE III-6

LOX/LH_2, P_c = 1,000

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

LOX/H2 JAN 5,1960

PC= 1000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	.691	.556	1.247	1.232	310.3	306.7	4465	3991
4.00	7973	.677	.564	1.241	1.226	307.6	303.9	5360	4899
5.00	7818	.667	.571	1.237	1.222	300.6	297.0	5966	5562
6.00	7595	.659	.575	1.234	1.219	291.3	287.8	6322	5972
7.00	7334	.655	.577	1.232	1.218	280.9	277.6	6478	6159
9.00	6820	.654	.578	1.232	1.217	261.2	258.1	6456	6142

PC= 1000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.237	.225	1.462	1.433	363.8	356.5	4465	2867
4.00	7973	1.228	.237	1.466	1.436	363.3	356.0	5360	3708
5.00	7818	1.218	.252	1.470	1.441	357.3	350.2	5966	4471
6.00	7595	1.210	.264	1.474	1.445	348.0	341.1	6322	5057
7.00	7334	1.213	.269	1.482	1.452	337.8	331.1	6478	5365
9.00	6820	1.205	.272	1.477	1.448	313.1	306.9	6456	5382

PC= 1000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.364	.178	1.542	1.498	383.7	372.8	4465	2488
4.00	7973	1.358	.191	1.550	1.506	384.1	373.2	5360	3290
5.00	7818	1.355	.205	1.560	1.516	379.1	368.3	5966	4043
6.00	7595	1.353	.203	1.557	1.513	367.5	357.1	6322	4686
7.00	7334	1.357	.209	1.566	1.522	357.0	347.0	6478	5072
9.00	6820	1.349	.214	1.562	1.518	331.2	321.9	6456	5104

PC= 1000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.436	.152	1.588	1.529	395.1	380.5	4465	2251
4.00	7973	1.435	.161	1.596	1.537	395.5	380.9	5360	3019
5.00	7818	1.435	.174	1.609	1.550	391.0	376.7	5966	3759
6.00	7595	1.433	.189	1.622	1.563	382.9	369.0	6322	4429
7.00	7334	1.431	.200	1.630	1.572	371.7	358.3	6478	4888
9.00	6820	1.430	.201	1.632	1.573	346.0	333.5	6456	4921

PC= 1000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.484	.133	1.617	1.544	402.5	384.2	4465	2083
4.00	7973	1.485	.146	1.631	1.558	404.3	386.1	5360	2826
5.00	7818	1.487	.160	1.648	1.574	400.4	382.6	5966	3551
6.00	7595	1.489	.175	1.663	1.590	392.8	375.4	6322	4231
7.00	7334	1.488	.186	1.674	1.601	381.6	364.9	6478	4737
9.00	6820	1.483	.189	1.673	1.599	354.6	339.0	6456	4794

PC= 1000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.518	.124	1.642	1.554	408.8	386.8	4465	1957
4.00	7973	1.521	.138	1.659	1.571	411.2	389.3	5360	2678
5.00	7818	1.526	.151	1.678	1.589	407.7	386.3	5966	3389
6.00	7595	1.532	.160	1.692	1.603	399.4	378.6	6322	4071
7.00	7334	1.532	.173	1.705	1.616	388.6	368.5	6478	4611
9.00	6820	1.526	.177	1.703	1.615	361.0	342.3	6456	4684

TABLE III-6

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

LOX/H2 JAN 5,1960

PC= 1000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.545	.114	1.659	1.557	413.0	387.4	4465	1853
4.00	7973	1.551	.127	1.678	1.575	415.8	390.3	5360	2555
5.00	7818	1.558	.141	1.699	1.596	412.8	387.8	5966	3255
6.00	7595	1.564	.155	1.719	1.616	405.8	381.5	6322	3938
7.00	7334	1.566	.168	1.734	1.631	395.3	371.9	6478	4504
9.00	6820	1.565	.172	1.736	1.633	368.1	346.3	6456	4583

PC= 1000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.567	.109	1.675	1.558	417.0	387.7	4465	1769
4.00	7973	1.574	.122	1.695	1.578	420.2	391.1	5360	2455
5.00	7818	1.584	.132	1.716	1.598	416.9	388.3	5966	3142
6.00	7595	1.591	.146	1.737	1.620	410.2	382.4	6322	3823
7.00	7334	1.595	.160	1.755	1.637	400.0	373.2	6478	4410
9.00	6820	1.594	.163	1.758	1.640	372.6	347.7	6456	4502

PC= 1000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.600	.098	1.698	1.551	422.7	386.1	4465	1635
4.00	7973	1.610	.111	1.721	1.574	426.6	390.2	5360	2292
5.00	7818	1.624	.122	1.746	1.599	424.3	388.5	5966	2959
6.00	7595	1.633	.136	1.770	1.623	417.8	383.1	6322	3636
7.00	7334	1.638	.150	1.788	1.642	407.7	374.2	6478	4256
9.00	6820	1.635	.155	1.790	1.643	379.5	348.3	6456	4380

PC= 1000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.625	.091	1.716	1.540	427.1	383.2	4465	1531
4.00	7973	1.637	.102	1.739	1.563	431.0	387.3	5360	2171
5.00	7818	1.657	.113	1.770	1.594	430.2	387.3	5966	2799
6.00	7595	1.666	.128	1.794	1.618	423.6	382.0	6322	3481
7.00	7334	1.675	.141	1.816	1.639	413.9	373.7	6478	4114
9.00	6820	1.670	.146	1.816	1.640	385.1	347.7	6456	4271

PC= 1000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.652	.083	1.735	1.515	431.8	376.9	4465	1415
4.00	7973	1.665	.095	1.759	1.539	436.1	381.4	5360	2039
5.00	7818	1.686	.106	1.792	1.572	435.5	382.0	5966	2655
6.00	7595	1.702	.119	1.821	1.601	430.0	377.9	6322	3310
7.00	7334	1.718	.130	1.849	1.628	421.5	371.2	6478	3932
9.00	6820	1.714	.137	1.851	1.631	392.4	345.7	6456	4125

PC= 1000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.684	.073	1.757	1.463	437.3	364.2	4465	1276
4.00	7973	1.704	.084	1.788	1.494	443.1	370.2	5360	1846
5.00	7818	1.725	.096	1.821	1.527	442.4	371.0	5966	2454
6.00	7595	1.749	.107	1.857	1.563	438.3	368.9	6322	3067
7.00	7334	1.761	.121	1.882	1.588	429.0	362.0	6478	3737
9.00	6820	1.761	.128	1.889	1.595	400.4	338.1	6456	3952

TABLE III-6

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

LOX/H2 JAN 5,1960

PC= 1000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000043000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.722	.062	1.784	1.343	444.0	334.3	4465	1101
4.00	7973	1.747	.072	1.819	1.378	450.9	341.6	5360	1618
5.00	7818	1.774	.083	1.857	1.416	451.3	344.1	5966	2189
6.00	7595	1.800	.095	1.894	1.453	447.3	343.2	6322	2794
7.00	7334	1.820	.108	1.928	1.488	439.6	339.1	6478	3438
9.00	6820	1.823	.115	1.938	1.497	410.9	317.4	6456	3696

PC= 1000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.746	.055	1.801	1.213	448.1	301.8	4465	990
4.00	7973	1.774	.065	1.839	1.251	455.8	310.1	5360	1471
5.00	7818	1.804	.075	1.879	1.292	456.7	313.9	5966	2014
6.00	7595	1.834	.087	1.921	1.333	453.5	314.7	6322	2599
7.00	7334	1.863	.099	1.961	1.374	447.1	313.1	6478	3204
9.00	6820	1.863	.107	1.970	1.382	417.7	293.1	6456	3512

PC= 1000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
3.00	8007	1.762	.050	1.812	1.077	451.0	268.1	4465	913
4.00	7973	1.793	.059	1.852	1.117	459.1	276.9	5360	1364
5.00	7818	1.828	.069	1.897	1.162	460.9	282.4	5966	1877
6.00	7595	1.858	.081	1.940	1.205	458.0	284.5	6322	2454
7.00	7334	1.886	.093	1.979	1.244	451.2	283.6	6478	3073
9.00	6820	1.892	.101	1.993	1.258	422.6	266.8	6456	3366

TABLE III-7

LOX/LH_2, $P_c = 2,000$

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 2000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	.678	.565	1.242	1.235	308.1	306.3	5398	4921
5.00	7842	.668	.570	1.238	1.231	301.8	300.0	6055	5623
6.00	7639	.662	.573	1.234	1.227	293.1	291.3	6461	6082
7.00	7385	.653	.580	1.233	1.226	283.0	281.4	6647	6308

PC= 2000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.228	.238	1.465	1.451	363.4	359.8	5398	3711
5.00	7842	1.219	.250	1.469	1.454	358.1	354.5	6055	4476
6.00	7639	1.210	.262	1.472	1.457	349.6	346.1	6461	5094
7.00	7385	1.205	.270	1.475	1.460	338.6	335.3	6647	5466

PC= 2000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.360	.190	1.550	1.528	384.4	379.0	5398	3285
5.00	7842	1.355	.203	1.558	1.536	379.8	374.5	6055	4036
6.00	7639	1.353	.201	1.554	1.532	369.0	363.8	6461	4698
7.00	7385	1.349	.211	1.560	1.538	358.1	353.0	6647	5148

PC= 2000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.436	.160	1.596	1.566	395.8	388.5	5398	3015
5.00	7842	1.434	.173	1.607	1.578	391.8	384.6	6055	3747
6.00	7639	1.432	.186	1.618	1.589	384.3	377.3	6461	4425
7.00	7385	1.430	.198	1.628	1.599	373.7	367.0	6647	4934

PC= 2000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.486	.146	1.631	1.595	404.6	395.5	5398	2821
5.00	7842	1.487	.159	1.646	1.609	401.1	392.2	6055	3538
6.00	7639	1.488	.172	1.659	1.623	394.1	385.4	6461	4219
7.00	7385	1.487	.184	1.671	1.635	383.7	375.3	6647	4768

PC= 2000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.522	.137	1.659	1.615	411.5	400.5	5398	2673
5.00	7842	1.526	.150	1.675	1.631	408.4	397.6	6055	3375
6.00	7639	1.530	.158	1.687	1.643	400.7	390.2	6461	4053
7.00	7385	1.531	.171	1.701	1.657	390.6	380.4	6647	4631

PC= 2000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.551	.127	1.678	1.626	416.1	403.4	5398	2551
5.00	7842	1.557	.139	1.696	1.645	413.4	400.9	6055	3241
6.00	7639	1.562	.152	1.714	1.663	407.1	394.8	6461	3916
7.00	7385	1.564	.166	1.730	1.679	397.2	385.4	6647	4516

PC= 2000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.574	.121	1.696	1.637	420.5	405.9	5398	2451
5.00	7842	1.582	.130	1.713	1.654	417.5	403.2	6055	3128
6.00	7639	1.589	.143	1.732	1.674	411.4	397.4	6461	3799
7.00	7385	1.593	.157	1.750	1.692	401.8	388.3	6647	4414

TABLE III-7 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 2000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.611	.111	1.721	1.648	426.8	408.6	5398	2290
5.00	7842	1.621	.121	1.742	1.669	424.7	406.8	6055	2949
6.00	7639	1.631	.134	1.764	1.691	419.0	401.5	6461	3609
7.00	7385	1.637	.148	1.785	1.711	409.7	392.9	6647	4242

PC= 2000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.638	.102	1.740	1.651	431.4	409.6	5398	2164
5.00	7842	1.650	.113	1.764	1.676	429.9	408.5	6055	2808
6.00	7639	1.662	.126	1.788	1.700	424.6	403.7	6461	3458
7.00	7385	1.672	.138	1.810	1.722	415.5	395.2	6647	4100

PC= 2000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.668	.094	1.762	1.652	437.0	409.6	5398	2018
5.00	7842	1.683	.105	1.789	1.678	436.0	409.1	6055	2644
6.00	7639	1.698	.117	1.815	1.705	431.1	404.9	6461	3280
7.00	7385	1.711	.129	1.840	1.730	422.3	397.0	6647	3925

PC= 2000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.704	.084	1.788	1.641	443.3	406.9	5398	1842
5.00	7842	1.722	.095	1.817	1.670	442.9	407.1	6055	2442
6.00	7639	1.741	.106	1.847	1.700	438.6	403.7	6461	3060
7.00	7385	1.757	.119	1.876	1.729	430.6	396.9	6647	3701

PC= 2000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.747	.071	1.819	1.599	451.1	396.4	5398	1613
5.00	7842	1.771	.082	1.853	1.633	451.7	398.0	6055	2178
6.00	7639	1.794	.093	1.887	1.667	448.1	395.8	6461	2768
7.00	7385	1.816	.106	1.922	1.701	441.2	390.6	6647	3394

PC= 2000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.774	.064	1.839	1.545	456.0	383.1	5398	1466
5.00	7842	1.801	.074	1.875	1.581	457.1	385.5	6055	2004
6.00	7639	1.828	.085	1.913	1.619	454.3	384.5	6461	2574
7.00	7385	1.854	.097	1.951	1.657	447.9	380.4	6647	3184

PC= 2000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7978	1.793	.058	1.852	1.485	459.3	368.2	5398	1359
5.00	7842	1.823	.069	1.892	1.524	461.1	371.5	6055	1876
6.00	7639	1.852	.080	1.932	1.564	458.7	371.4	6461	2431
7.00	7385	1.881	.091	1.972	1.604	452.6	368.3	6647	3027

TABLE III-8

LOX/LH_2, $P_c = 4,000$

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 4000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	.684	.559	1.243	1.239	308.4	307.5	5428	4928
5.00	7859	.668	.571	1.239	1.235	302.7	301.8	6132	5675
6.00	7674	.666	.569	1.235	1.231	294.7	293.8	6591	6178
7.00	7433	.658	.576	1.234	1.230	285.0	284.2	6814	6446

PC= 4000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.229	.237	1.466	1.458	363.7	361.9	5428	3708
5.00	7859	1.220	.249	1.469	1.462	358.9	357.1	6132	4474
6.00	7674	1.212	.260	1.472	1.464	351.1	349.3	6591	5119
7.00	7433	1.206	.268	1.475	1.467	340.7	339.1	6814	5543

PC= 4000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.360	.190	1.550	1.539	384.6	381.9	5428	3282
5.00	7859	1.356	.202	1.557	1.546	380.5	377.8	6132	4027
6.00	7674	1.351	.213	1.564	1.553	373.2	370.6	6591	4699
7.00	7433	1.350	.209	1.559	1.548	360.2	357.6	6814	5199

PC= 4000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.436	.160	1.596	1.581	395.9	392.3	5428	3011
5.00	7859	1.435	.171	1.606	1.591	392.4	388.8	6132	3736
6.00	7674	1.433	.183	1.616	1.601	385.6	382.1	6591	4415
7.00	7433	1.431	.195	1.626	1.611	375.7	372.3	6814	4965

PC= 4000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.486	.146	1.631	1.613	404.7	400.2	5428	2818
5.00	7859	1.487	.157	1.644	1.626	401.7	397.2	6132	3526
6.00	7674	1.487	.169	1.657	1.638	395.3	390.9	6591	4202
7.00	7433	1.487	.182	1.669	1.651	385.7	381.4	6814	4784

PC= 4000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.522	.137	1.659	1.637	411.6	406.1	5428	2670
5.00	7859	1.525	.148	1.674	1.652	408.9	403.5	6132	3364
6.00	7674	1.529	.155	1.684	1.662	401.8	396.6	6591	4033
7.00	7433	1.531	.168	1.699	1.677	392.5	387.4	6814	4636

PC= 4000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.551	.126	1.678	1.652	416.2	409.8	5428	2548
5.00	7859	1.556	.138	1.694	1.668	413.9	407.6	6132	3229
6.00	7674	1.561	.150	1.711	1.685	408.1	402.0	6591	3895
7.00	7433	1.564	.163	1.727	1.701	399.1	393.1	6814	4512

PC= 4000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.574	.121	1.695	1.666	420.6	413.3	5428	2448
5.00	7859	1.581	.129	1.711	1.681	418.0	410.8	6132	3117
6.00	7674	1.588	.141	1.729	1.699	412.4	405.4	6591	3776
7.00	7433	1.592	.154	1.747	1.717	403.6	396.8	6814	4403

TABLE III-8 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 4000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.610	.110	1.721	1.684	426.9	417.8	5428	2287
5.00	7859	1.620	.120	1.740	1.703	425.1	416.1	6132	2938
6.00	7674	1.629	.132	1.760	1.724	419.9	411.2	6591	3585
7.00	7433	1.636	.145	1.781	1.744	411.5	403.0	6814	4220

PC= 4000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.638	.101	1.739	1.695	431.5	420.6	5428	2161
5.00	7859	1.649	.113	1.761	1.717	430.3	419.6	6132	2797
6.00	7674	1.660	.124	1.784	1.740	425.5	415.0	6591	3434
7.00	7433	1.670	.137	1.806	1.762	417.4	407.2	6814	4072

PC= 4000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.668	.093	1.761	1.706	437.0	423.4	5428	2016
5.00	7859	1.682	.104	1.786	1.731	436.3	422.9	6132	2633
6.00	7674	1.695	.116	1.811	1.756	432.0	418.8	6591	3257
7.00	7433	1.708	.126	1.835	1.779	423.9	411.2	6814	3892

PC= 4000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.704	.084	1.787	1.714	443.4	425.2	5428	1839
5.00	7859	1.720	.094	1.814	1.741	443.3	425.3	6132	2432
6.00	7674	1.737	.105	1.842	1.768	439.4	421.9	6591	3037
7.00	7433	1.754	.116	1.870	1.796	432.1	415.1	6814	3664

PC= 4000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.747	.071	1.818	1.708	451.2	423.8	5428	1611
5.00	7859	1.768	.082	1.850	1.740	452.0	425.1	6132	2169
6.00	7674	1.790	.092	1.881	1.771	448.8	422.5	6591	2746
7.00	7433	1.812	.103	1.915	1.805	442.5	417.0	6814	3355

PC= 4000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.774	.064	1.838	1.691	456.1	419.6	5428	1464
5.00	7859	1.799	.074	1.872	1.725	457.4	421.5	6132	1995
6.00	7674	1.823	.084	1.907	1.760	454.9	419.8	6591	2553
7.00	7433	1.849	.095	1.943	1.796	449.1	415.1	6814	3145

PC= 4000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7982	1.793	.058	1.851	1.668	459.4	413.8	5428	1357
5.00	7859	1.820	.068	1.888	1.705	461.3	416.4	6132	1868
6.00	7674	1.847	.079	1.925	1.741	459.3	415.4	6591	2411
7.00	7433	1.875	.089	1.964	1.780	453.7	411.3	6814	2990

TABLE III-9

LOX/LH_2, $P_c = 6,000$

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 6000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	.684	.559	1.243	1.241	308.6	308.0	5442	4934
5.00	7871	.667	.572	1.239	1.237	303.2	302.6	6172	5702
6.00	7693	.654	.581	1.236	1.233	295.5	294.9	6662	6248
7.00	7461	.653	.581	1.234	1.231	286.1	285.6	6910	6533

PC= 6000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.229	.237	1.466	1.461	363.8	362.6	5442	3707
5.00	7871	1.221	.248	1.468	1.463	359.2	358.0	6172	4473
6.00	7693	1.213	.258	1.471	1.466	351.8	350.7	6662	5130
7.00	7461	1.207	.267	1.474	1.469	341.9	340.8	6910	5583

PC= 6000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.360	.190	1.550	1.543	384.7	382.9	5442	3280
5.00	7871	1.356	.201	1.557	1.549	380.8	379.0	6172	4022
6.00	7693	1.351	.212	1.564	1.556	373.9	372.1	6662	4698
7.00	7461	1.351	.207	1.558	1.551	361.3	359.6	6910	5222

PC= 6000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.435	.164	1.598	1.589	396.7	394.3	5442	3010
5.00	7871	1.434	.170	1.605	1.595	392.6	390.3	6172	3730
6.00	7693	1.433	.182	1.615	1.605	386.2	383.9	6662	4408
7.00	7461	1.431	.194	1.625	1.615	376.8	374.6	6910	4977

PC= 6000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.485	.145	1.631	1.619	404.8	401.8	5442	2816
5.00	7871	1.486	.156	1.643	1.631	402.0	399.0	6172	3519
6.00	7693	1.487	.168	1.656	1.643	395.9	393.0	6662	4193
7.00	7461	1.487	.180	1.668	1.655	386.7	383.9	6910	4789

PC= 6000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.522	.137	1.658	1.644	411.7	408.0	5442	2668
5.00	7871	1.524	.148	1.672	1.657	409.1	405.5	6172	3357
6.00	7693	1.529	.154	1.683	1.668	402.4	398.9	6662	4022
7.00	7461	1.531	.166	1.697	1.682	393.5	390.1	6910	4635

PC= 6000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.551	.126	1.677	1.660	416.3	412.0	5442	2546
5.00	7871	1.555	.137	1.693	1.676	414.1	409.9	6172	3223
6.00	7693	1.560	.149	1.709	1.692	408.7	404.6	6662	3883
7.00	7461	1.563	.162	1.725	1.708	400.1	396.1	6910	4506

PC= 6000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.574	.121	1.695	1.675	420.7	415.8	5442	2446
5.00	7871	1.580	.129	1.709	1.690	418.2	413.4	6172	3111
6.00	7693	1.587	.140	1.727	1.707	412.9	408.2	6662	3764
7.00	7461	1.592	.153	1.745	1.725	404.6	400.0	6910	4393

TABLE III-9 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 1 18 NOV, 60

O2(LIQUID)/H2(LIQUID) MAY 27,1959

PC= 6000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.610	.110	1.720	1.696	427.0	420.9	5442	2285
5.00	7871	1.619	.120	1.738	1.714	425.3	419.3	6172	2932
6.00	7693	1.627	.131	1.758	1.734	420.4	414.6	6662	3572
7.00	7461	1.635	.143	1.778	1.754	412.4	406.7	6910	4206

PC= 6000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.637	.101	1.739	1.709	431.6	424.3	5442	2160
5.00	7871	1.648	.112	1.760	1.730	430.5	423.3	6172	2792
6.00	7693	1.658	.123	1.781	1.752	426.0	418.9	6662	3422
7.00	7461	1.668	.135	1.804	1.774	418.3	411.5	6910	4055

PC= 6000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.668	.093	1.761	1.724	437.1	428.0	5442	2015
5.00	7871	1.680	.104	1.784	1.747	436.5	427.5	6172	2628
6.00	7693	1.693	.115	1.808	1.771	432.4	423.6	6662	3245
7.00	7461	1.707	.125	1.831	1.795	424.8	416.2	6910	3872

PC= 6000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.703	.084	1.787	1.738	443.5	431.3	5442	1838
5.00	7871	1.719	.094	1.812	1.763	443.4	431.4	6172	2427
6.00	7693	1.735	.104	1.839	1.790	439.8	428.1	6662	3025
7.00	7461	1.752	.115	1.866	1.817	432.9	421.5	6910	3643

PC= 6000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.747	.071	1.818	1.744	451.2	433.0	5442	1610
5.00	7871	1.766	.081	1.848	1.774	452.1	434.2	6172	2165
6.00	7693	1.787	.091	1.878	1.805	449.2	431.6	6662	2735
7.00	7461	1.809	.102	1.911	1.837	443.2	426.1	6910	3334

PC= 6000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.773	.064	1.838	1.740	456.1	431.8	5442	1463
5.00	7871	1.797	.073	1.870	1.772	457.5	433.5	6172	1991
6.00	7693	1.820	.083	1.904	1.806	455.2	431.8	6662	2543
7.00	7461	1.845	.094	1.939	1.841	449.7	427.0	6910	3125

PC= 6000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
4.00	7985	1.792	.058	1.851	1.728	459.4	429.0	5442	1356
5.00	7871	1.818	.068	1.886	1.763	461.4	431.5	6172	1863
6.00	7693	1.844	.078	1.922	1.799	459.6	430.3	6662	2401
7.00	7461	1.872	.087	1.959	1.837	454.3	425.9	6910	2970

TABLE III-10

LOX/LH_2, P_c = 10,000

ENGINE DESIGN HANDBOOK DATA. TAPE 8A 05 JUNE 1961

LO2/LH2 SEPT.2,1960

PC=10000 AE/AT= 1ATMOS ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	11704	.049	1.754	1.704	426.3	414.3	3292	625
4.00	7990	1.778	.062	1.840	1.778	457.0	441.5	5455	1433
6.00	7714	1.844	.076	1.920	1.844	460.5	442.2	6745	2365
8.00	7232	1.910	.093	2.002	1.910	450.2	429.4	7082	3357
10.00	6731	1.889	.086	1.975	1.889	413.2	395.2	6873	2900

PC=10000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	.704	.550	1.254	1.252	304.8	304.4	3292	2879
4.00	7990	.686	.557	1.243	1.242	308.8	308.4	5455	4938
6.00	7714	.663	.573	1.236	1.235	296.4	296.0	6745	6299
8.00	7232	.658	.576	1.233	1.232	277.3	277.0	7082	6693
10.00	6731	.656	.578	1.234	1.232	258.2	257.8	6873	6479

PC=10000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.370	.162	1.532	1.528	372.5	371.4	3292	1640
4.00	7990	1.360	.189	1.549	1.545	384.7	383.6	5455	3278
6.00	7714	1.353	.210	1.563	1.559	374.9	373.8	6745	4688
8.00	7232	1.350	.209	1.560	1.555	350.6	349.6	7082	5430
10.00	6731	1.351	.206	1.558	1.553	325.9	325.0	6873	5139

PC=10000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.438	.137	1.575	1.569	382.8	381.3	3292	1459
4.00	7992	1.434	.163	1.597	1.592	396.8	395.3	5455	3009
6.00	7714	1.434	.180	1.614	1.608	387.0	385.6	6745	4394
8.00	7232	1.430	.197	1.627	1.622	365.9	364.6	7082	5214
10.00	6731	1.431	.193	1.624	1.618	339.8	338.6	6873	4898

PC=10000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.482	.122	1.604	1.597	389.9	388.2	3292	1334
4.00	7990	1.485	.145	1.630	1.622	404.8	403.0	5455	2815
6.00	7714	1.487	.166	1.654	1.646	396.6	394.8	6745	4178
8.00	7232	1.487	.184	1.671	1.664	375.7	374.0	7082	5052
10.00	6731	1.487	.180	1.667	1.659	348.7	347.2	6873	4718

PC=10000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.515	.109	1.624	1.615	394.7	392.6	3292	1239
4.00	7990	1.521	.137	1.657	1.649	411.7	409.5	5455	2667
6.00	7714	1.528	.153	1.681	1.672	403.0	400.9	6745	4006
8.00	7232	1.530	.171	1.701	1.692	382.4	380.5	7082	4922
10.00	6731	1.530	.166	1.696	1.687	354.8	353.0	6873	4572

PC=10000 AE/AT2 7.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.539	.102	1.641	1.631	398.9	396.4	3292	1165
4.00	7990	1.550	.126	1.676	1.666	416.3	413.7	5455	2545
6.00	7714	1.559	.147	1.707	1.696	409.2	406.8	6745	3867
8.00	7232	1.563	.167	1.730	1.720	389.0	386.7	7082	4817
10.00	6731	1.562	.162	1.724	1.714	360.8	358.6	6873	4452

TABLE III-10 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 8A 05 JUNE 1961

LO2/LH2 SEPT.2,1960

PC=10000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.559	.095	1.654	1.642	402.0	399.1	3292	1104
4.00	7990	1.573	.121	1.694	1.682	420.7	417.8	5455	2445
6.00	7714	1.586	.139	1.724	1.712	413.5	410.6	6745	3748
8.00	7232	1.592	.159	1.750	1.739	393.6	390.9	7082	4724
10.00	6731	1.590	.153	1.744	1.732	364.8	362.4	6873	4347

PC=10000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.589	.085	1.674	1.659	406.8	403.3	3292	1011
4.00	7990	1.609	.110	1.719	1.704	427.0	423.4	5455	2284
6.00	7714	1.626	.129	1.755	1.741	420.9	417.4	6745	3557
8.00	7232	1.635	.150	1.785	1.771	401.4	398.1	7082	4575
10.00	6731	1.634	.144	1.777	1.763	371.9	368.8	6873	4174

PC=10000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.611	.078	1.689	1.671	410.6	406.3	3292	940
4.00	7990	1.636	.101	1.738	1.720	431.6	427.2	5455	2158
6.00	7714	1.657	.122	1.779	1.761	426.5	422.3	6745	3404
8.00	7232	1.669	.141	1.811	1.793	407.1	403.1	7082	4452
10.00	6731	1.667	.136	1.803	1.785	377.2	373.5	6873	4035

PC=10000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.635	.071	1.705	1.683	414.6	409.2	3292	861
4.00	7990	1.667	.093	1.760	1.738	437.1	431.6	5455	2012
6.00	7714	1.691	.114	1.805	1.783	432.8	427.5	6745	3230
8.00	7232	1.708	.133	1.841	1.819	414.0	409.0	7082	4304
10.00	6731	1.705	.126	1.831	1.809	383.1	378.5	6873	3866

PC=10000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.663	.062	1.725	1.695	419.2	412.1	3292	768
4.00	7990	1.702	.084	1.785	1.756	443.4	436.1	5455	1838
6.00	7714	1.733	.103	1.836	1.806	440.2	433.1	6745	3011
8.00	7232	1.755	.124	1.879	1.849	422.4	415.8	7082	4114
10.00	6731	1.750	.116	1.866	1.837	390.5	384.4	6873	3651

PC=10000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.696	.052	1.747	1.703	424.8	414.1	3292	654
4.00	7990	1.745	.071	1.816	1.772	451.2	440.2	5455	1611
6.00	7714	1.784	.090	1.874	1.830	449.5	438.9	6745	2723
8.00	7232	1.815	.112	1.927	1.883	433.2	423.3	7082	3846
10.00	6731	1.808	.104	1.912	1.867	400.0	390.7	6873	3358

PC=10000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.716	.046	1.761	1.703	428.2	413.9	3292	584
4.00	7990	1.772	.064	1.836	1.777	456.1	441.5	5455	1464
6.00	7714	1.817	.083	1.899	1.841	455.5	441.4	6745	2532
8.00	7232	1.854	.104	1.958	1.899	440.2	427.0	7082	3656
10.00	6731	1.845	.096	1.940	1.881	406.0	393.7	6873	3156

TABLE III-10 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 8A 05 JUNE 1961

LO2/LH2 SEPT.2,1960

PC=10000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000040000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	7820	1.729	.044	1.773	1.699	431.0	413.1	3292	537
4.00	7990	1.791	.058	1.849	1.776	459.3	441.1	5455	1357
6.00	7714	1.840	.077	1.918	1.844	459.8	442.2	6745	2389
8.00	7232	1.882	.098	1.980	1.907	445.3	428.7	7082	3509
10.00	6731	1.871	.089	1.961	1.887	410.2	394.8	6873	3004

TABLE III-11

LOX/C_3H_8, P_c = 300

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 300 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	.675	.565	1.240	1.191	227.2	218.2	5441	4998
2.40	5994	.660	.575	1.234	1.185	229.9	220.8	6019	5679
2.80	5924	.652	.579	1.231	1.182	226.8	217.8	6219	5930
3.20	5806	.651	.580	1.230	1.181	222.1	213.2	6262	5988
3.60	5684	.650	.580	1.230	1.182	217.4	208.8	6244	5976
4.00	5571	.650	.581	1.230	1.181	213.0	204.5	6198	5932
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.39	5993	.660	.574	1.234	1.185	230.0	220.8	6005	5663

PC= 300 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.225	.239	1.465	1.367	268.3	250.4	5441	3811
2.40	5994	1.212	.260	1.472	1.374	274.3	256.1	6019	4741
2.80	5924	1.202	.273	1.475	1.377	271.6	253.6	6219	5222
3.20	5806	1.200	.276	1.476	1.379	266.5	248.8	6262	5342
3.60	5684	1.198	.278	1.476	1.378	260.8	243.5	6244	5348
4.00	5571	1.198	.277	1.476	1.378	255.6	238.6	6198	5307
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.48	5994	1.210	.263	1.473	1.375	274.5	256.2	6086	4867

PC= 300 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.358	.192	1.550	1.403	283.9	256.9	5441	3383
2.40	5994	1.352	.213	1.565	1.418	291.5	264.2	6019	4346
2.80	5924	1.346	.215	1.560	1.413	287.3	260.3	6219	4953
3.20	5806	1.346	.218	1.564	1.417	282.3	255.7	6262	5113
3.60	5684	1.345	.219	1.564	1.417	276.3	250.3	6244	5127
4.00	5571	1.345	.219	1.564	1.417	270.8	245.3	6198	5087
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	5995	1.351	.215	1.565	1.418	291.7	264.3	6072	4457

PC= 300 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.434	.162	1.596	1.400	292.3	256.4	5441	3113
2.40	5994	1.434	.183	1.616	1.420	301.1	264.6	6019	4074
2.80	5924	1.428	.202	1.630	1.434	300.2	264.1	6219	4767
3.20	5806	1.428	.207	1.635	1.439	295.1	259.7	6262	4967
3.60	5684	1.426	.208	1.635	1.439	288.8	254.2	6244	4989
4.00	5571	1.426	.208	1.635	1.439	283.1	249.1	6198	4948
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.56	5987	1.432	.190	1.623	1.427	301.9	265.5	6145	4386

PC= 300 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.484	.148	1.632	1.387	298.9	254.0	5441	2923
2.40	5994	1.488	.168	1.657	1.412	308.7	263.0	6019	3868
2.80	5924	1.487	.188	1.675	1.430	308.5	263.4	6219	4620
3.20	5806	1.486	.194	1.680	1.435	303.2	258.9	6262	4861
3.60	5684	1.484	.196	1.680	1.435	296.8	253.6	6244	4887
4.00	5571	1.484	.196	1.680	1.435	290.9	248.5	6198	4846
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.59	5982	1.488	.178	1.666	1.421	309.8	264.3	6163	4256

TABLE III-11 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.2,1960

PC= 300 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.520	.140	1.660	1.366	304.0	250.2	5441	2780
2.40	5994	1.530	.154	1.684	1.390	313.7	259.0	6019	3704
2.80	5924	1.532	.174	1.706	1.412	314.2	260.0	6219	4499
3.20	5806	1.530	.181	1.712	1.418	308.9	255.9	6262	4775
3.60	5684	1.530	.183	1.713	1.419	302.6	250.6	6244	4806
4.00	5571	1.529	.183	1.712	1.418	296.5	245.6	6198	4764
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.63	5960	1.531	.167	1.698	1.405	314.7	260.2	6153	4225

PC= 300 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.549	.130	1.679	1.336	307.5	244.7	5441	2663
2.40	5994	1.561	.148	1.710	1.367	318.6	254.7	6019	3571
2.80	5924	1.564	.170	1.734	1.391	319.3	256.2	6219	4400
3.20	5806	1.564	.178	1.742	1.399	314.4	252.5	6262	4706
3.60	5684	1.565	.179	1.744	1.401	308.1	247.5	6244	4740
4.00	5571	1.564	.179	1.743	1.400	301.9	242.5	6198	4699
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.65	5956	1.563	.164	1.727	1.384	319.7	256.2	6163	4154

PC= 300 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.573	.124	1.697	1.305	310.8	239.1	5441	2568
2.40	5994	1.588	.139	1.728	1.336	321.9	248.9	6019	3457
2.80	5924	1.593	.161	1.754	1.362	323.1	250.9	6219	4308
3.20	5806	1.594	.170	1.764	1.372	318.3	247.6	6262	4646
3.60	5684	1.594	.172	1.766	1.374	312.0	242.8	6244	4684
4.00	5571	1.594	.172	1.766	1.374	305.8	237.9	6198	4642
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.68	5950	1.592	.156	1.748	1.356	323.4	250.9	6176	4108

PC= 300 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.609	.114	1.723	1.233	315.6	225.9	5441	2418
2.40	5994	1.629	.130	1.759	1.269	327.7	236.4	6019	3275
2.80	5924	1.640	.151	1.791	1.301	329.8	239.6	6219	4145
3.20	5806	1.639	.161	1.800	1.310	324.9	236.5	6262	4549
3.60	5684	1.640	.163	1.803	1.313	318.5	232.0	6244	4595
4.00	5571	1.640	.163	1.803	1.313	312.2	227.3	6198	4551
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.72	5942	1.638	.148	1.786	1.296	329.9	239.4	6192	4009

PC= 300 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.637	.106	1.743	1.155	319.2	211.5	5441	2303
2.40	5994	1.660	.122	1.782	1.194	332.0	222.5	6019	3133
2.80	5924	1.675	.142	1.816	1.229	334.5	226.3	6219	4010
3.20	5806	1.674	.153	1.828	1.240	329.8	223.7	6262	4469
3.60	5684	1.675	.155	1.830	1.243	323.4	219.6	6244	4523
4.00	5571	1.675	.155	1.830	1.242	317.0	215.2	6198	4478
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.74	5938	1.674	.139	1.813	1.225	334.6	226.1	6199	3905

TABLE III-11 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.2,1960

PC= 300 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.668	.099	1.766	1.031	323.5	188.9	5441	2172
2.40	5994	1.695	.114	1.808	1.074	336.9	200.0	6019	2968
2.80	5924	1.714	.132	1.847	1.112	340.1	204.8	6219	3845
3.20	5806	1.715	.146	1.861	1.126	335.9	203.3	6262	4372
3.60	5684	1.718	.148	1.865	1.130	329.6	199.7	6244	4437
4.00	5571	1.717	.147	1.864	1.129	322.8	195.6	6198	4391
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.77	5931	1.714	.131	1.845	1.110	340.1	204.6	6210	3795

PC= 300 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.704	.090	1.793	.813	328.4	149.0	5441	2018
2.40	5994	1.736	.103	1.839	.859	342.7	160.1	6019	2768
2.80	5924	1.762	.122	1.884	.904	346.9	166.5	6219	3630
3.20	5806	1.766	.137	1.903	.923	343.4	166.6	6262	4246
3.60	5684	1.767	.140	1.906	.926	336.8	163.7	6244	4332
4.00	5571	1.767	.139	1.906	.926	330.1	160.4	6198	4281
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.82	5920	1.763	.122	1.885	.905	346.9	166.6	6225	3664

PC= 300 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.748	.079	1.828	.358	334.8	65.6	5441	1828
2.40	5994	1.787	.091	1.878	.408	349.9	76.1	6019	2512
2.80	5924	1.822	.108	1.930	.461	355.5	84.9	6219	3333
3.20	5806	1.830	.125	1.955	.486	352.9	87.6	6262	4064
3.60	5684	1.832	.129	1.960	.491	346.4	86.7	6244	4186
4.00	5571	1.831	.128	1.959	.490	339.3	84.8	6198	4131
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.87	5907	1.825	.112	1.937	.467	355.7	85.8	6238	3471

PC= 300 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.777	.073	1.849	.000	338.7	.0	5441	1708
2.40	5994	1.820	.083	1.903	.000	354.6	.0	6019	2347
2.80	5924	1.861	.100	1.960	.001	361.0	.2	6219	3131
3.20	5806	1.873	.117	1.990	.031	359.2	5.6	6262	3922
3.60	5684	1.874	.122	1.996	.036	352.6	6.4	6244	4087
4.00	5571	1.875	.121	1.995	.036	345.5	6.2	6198	4024
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.91	5896	1.867	.104	1.971	.006	361.3	1.1	6247	3353

PC= 300 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5892	1.797	.069	1.865	.000	341.6	.0	5441	1624
2.40	5994	1.843	.079	1.921	.000	358.0	.0	6019	2227
2.80	5924	1.888	.093	1.982	.000	364.9	.0	6219	2983
3.20	5806	1.904	.112	2.016	.000	363.8	.0	6262	3806
3.60	5684	1.906	.117	2.023	.000	357.4	.0	6244	4009
4.00	5571	1.905	.116	2.021	.000	350.0	.0	6198	3943
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.94	5888	1.897	.099	1.997	-.000	365.4	-.0	6252	3269

TABLE III-12

LOX/C_3H_8, P_c = 500

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP). NOV.8,1960

PC= 500 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	.677	.564	1.241	1.212	227.6	222.2	5485	5020
2.40	6018	.661	.574	1.235	1.205	230.9	225.4	6115	5752
2.80	5959	.653	.579	1.232	1.202	228.2	222.8	6346	6041
3.20	5842	.652	.580	1.231	1.202	223.6	218.2	6396	6108
3.60	5721	.651	.579	1.230	1.201	218.8	213.6	6377	6095
4.00	5605	.650	.580	1.231	1.201	214.4	209.3	6329	6051
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.42	6019	.660	.574	1.234	1.205	231.0	225.5	6136	5778

PC= 500 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.227	.238	1.465	1.406	268.6	257.8	5485	3806
2.40	6018	1.214	.258	1.471	1.412	275.2	264.2	6115	4752
2.80	5959	1.204	.272	1.476	1.417	273.4	262.5	6346	5288
3.20	5842	1.202	.275	1.477	1.418	268.2	257.5	6396	5429
3.60	5721	1.199	.276	1.476	1.417	262.4	252.0	6377	5437
4.00	5605	1.200	.277	1.476	1.417	257.2	247.0	6329	5395
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.51	6019	1.211	.262	1.473	1.414	275.6	264.6	6220	4944

PC= 500 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.358	.191	1.549	1.461	284.1	267.9	5485	3376
2.40	6018	1.352	.211	1.563	1.475	292.3	275.8	6115	4338
2.80	5959	1.350	.212	1.561	1.473	289.3	272.9	6346	4997
3.20	5842	1.347	.217	1.564	1.476	284.0	267.9	6396	5190
3.60	5721	1.345	.218	1.563	1.475	277.9	262.3	6377	5206
4.00	5605	1.345	.218	1.564	1.475	272.4	257.1	6329	5165
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.49	6020	1.351	.213	1.564	1.476	292.6	276.1	6203	4516

PC= 500 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.434	.161	1.595	1.478	292.5	270.9	5485	3106
2.40	6018	1.433	.180	1.614	1.496	301.9	279.9	6115	4058
2.80	5959	1.430	.199	1.630	1.512	301.9	280.1	6346	4799
3.20	5842	1.429	.206	1.635	1.517	296.8	275.5	6396	5037
3.60	5721	1.426	.207	1.634	1.516	290.5	269.6	6377	5061
4.00	5605	1.427	.207	1.634	1.517	284.8	264.3	6329	5019
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.60	5995	1.432	.192	1.623	1.506	302.5	280.6	6256	4499

PC= 500 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.484	.147	1.631	1.484	299.0	272.1	5485	2916
2.40	6018	1.488	.166	1.654	1.507	309.3	281.8	6115	3848
2.80	5959	1.488	.186	1.674	1.527	310.1	282.8	6346	4642
3.20	5842	1.485	.193	1.678	1.531	304.7	278.1	6396	4926
3.60	5721	1.484	.195	1.679	1.532	298.6	272.4	6377	4955
4.00	5605	1.485	.195	1.679	1.533	292.6	267.0	6329	4912
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.65	5988	1.488	.180	1.668	1.521	310.5	283.1	6280	4402

TABLE III-12 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 500 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.520	.139	1.659	1.483	304.2	271.8	5485	2774
2.40	6018	1.529	.152	1.680	1.504	314.3	281.4	6115	3682
2.80	5959	1.532	.172	1.704	1.527	315.6	283.0	6346	4512
3.20	5842	1.529	.181	1.709	1.533	310.4	278.4	6396	4837
3.60	5721	1.529	.182	1.711	1.535	304.3	272.9	6377	4870
4.00	5605	1.530	.182	1.712	1.535	298.2	267.5	6329	4827
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.68	5983	1.532	.167	1.699	1.522	315.9	283.1	6295	4314

PC= 500 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.549	.129	1.678	1.472	307.7	269.9	5485	2657
2.40	6018	1.560	.146	1.706	1.500	319.2	280.7	6115	3548
2.80	5959	1.564	.168	1.732	1.526	320.8	282.7	6346	4404
3.20	5842	1.563	.177	1.740	1.534	315.9	278.6	6396	4764
3.60	5721	1.564	.178	1.742	1.536	309.8	273.2	6377	4803
4.00	5605	1.564	.178	1.742	1.536	303.5	267.7	6329	4759
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.70	5979	1.564	.163	1.727	1.521	321.0	282.8	6306	4238

PC= 500 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.572	.124	1.696	1.461	311.0	267.9	5485	2562
2.40	6018	1.586	.137	1.724	1.489	322.4	278.4	6115	3433
2.80	5959	1.593	.159	1.752	1.517	324.5	281.0	6346	4305
3.20	5842	1.592	.169	1.762	1.527	319.9	277.2	6396	4700
3.60	5721	1.593	.171	1.764	1.529	313.7	271.9	6377	4744
4.00	5605	1.593	.171	1.764	1.529	307.4	266.4	6329	4700
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.72	5975	1.592	.156	1.748	1.513	324.6	281.0	6316	4177

PC= 500 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.608	.114	1.722	1.428	315.8	261.9	5485	2413
2.40	6018	1.626	.128	1.754	1.460	328.2	273.2	6115	3253
2.80	5959	1.639	.149	1.787	1.493	331.1	276.7	6346	4132
3.20	5842	1.638	.160	1.799	1.505	326.6	273.3	6396	4595
3.60	5721	1.638	.162	1.801	1.507	320.2	267.9	6377	4651
4.00	5605	1.639	.162	1.801	1.507	313.8	262.6	6329	4604
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.76	5968	1.638	.147	1.785	1.491	331.2	276.6	6331	4061

PC= 500 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.636	.106	1.741	1.389	319.3	254.6	5485	2298
2.40	6018	1.657	.120	1.777	1.425	332.5	266.5	6115	3110
2.80	5959	1.674	.139	1.813	1.460	335.8	270.5	6346	3991
3.20	5842	1.674	.152	1.826	1.474	331.6	267.6	6396	4510
3.60	5721	1.674	.154	1.828	1.476	325.1	262.4	6377	4576
4.00	5605	1.675	.154	1.829	1.476	318.7	257.2	6329	4528
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	5964	1.673	.138	1.811	1.459	335.8	270.4	6338	3952

TABLE III-12 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP). NOV.8,1960

PC= 500 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.667	.098	1.765	1.324	323.6	242.8	5485	2167
2.40	6018	1.691	.112	1.803	1.363	337.4	254.9	6115	2947
2.80	5959	1.713	.130	1.842	1.402	341.3	259.6	6346	3819
3.20	5842	1.715	.144	1.859	1.419	337.7	257.6	6396	4407
3.60	5721	1.716	.147	1.863	1.422	331.3	252.9	6377	4485
4.00	5605	1.716	.146	1.863	1.422	324.5	247.7	6329	4437
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.81	5958	1.713	.130	1.843	1.402	341.3	259.7	6349	3834

PC= 500 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.702	.089	1.792	1.204	328.5	220.7	5485	2014
2.40	6018	1.732	.102	1.834	1.246	343.0	233.1	6115	2748
2.80	5959	1.759	.119	1.879	1.291	348.0	239.1	6346	3599
3.20	5842	1.765	.136	1.900	1.313	345.1	238.3	6396	4271
3.60	5721	1.765	.138	1.903	1.316	338.5	234.0	6377	4375
4.00	5605	1.766	.138	1.904	1.316	331.8	229.3	6329	4321
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.85	5947	1.761	.121	1.883	1.295	348.1	239.4	6363	3696

PC= 500 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.747	.079	1.826	.944	334.8	173.1	5485	1824
2.40	6018	1.783	.089	1.872	.990	350.2	185.2	6115	2494
2.80	5959	1.818	.106	1.925	1.043	356.6	193.2	6346	3300
3.20	5842	1.830	.124	1.953	1.071	354.7	194.6	6396	4069
3.60	5721	1.830	.128	1.957	1.076	348.1	191.3	6377	4223
4.00	5605	1.830	.127	1.957	1.075	341.0	187.4	6329	4164
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.91	5933	1.824	.111	1.935	1.053	356.9	194.2	6377	3512

PC= 500 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.775	.072	1.848	.672	338.7	123.2	5485	1705
2.40	6018	1.814	.082	1.897	.721	354.8	134.9	6115	2330
2.80	5959	1.856	.098	1.954	.778	362.0	144.2	6346	3099
3.20	5842	1.871	.116	1.987	.811	360.7	147.2	6396	3917
3.60	5721	1.872	.120	1.992	.817	354.3	145.2	6377	4119
4.00	5605	1.873	.120	1.993	.817	347.2	142.4	6329	4050
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.94	5925	1.865	.104	1.968	.792	362.5	146.0	6384	3380

PC= 500 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5898	1.795	.068	1.864	.394	341.7	72.2	5485	1622
2.40	6018	1.838	.078	1.915	.446	358.3	83.3	6115	2211
2.80	5959	1.883	.091	1.975	.505	365.8	93.6	6346	2951
3.20	5842	1.902	.110	2.012	.543	365.4	98.5	6396	3787
3.60	5721	1.903	.116	2.019	.549	359.0	97.7	6377	4038
4.00	5605	1.904	.115	2.019	.549	351.7	95.7	6329	3965
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.98	5914	1.895	.099	1.994	.525	366.6	96.4	6391	3312

TABLE III-13

LOX/C_3H_8, P_c = 1,000

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.2,1960

PC= 1000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	.680	.562	1.242	1.227	228.0	225.3	5535	5044
2.40	6048	.663	.572	1.235	1.221	232.3	229.5	6242	5847
2.80	6004	.654	.579	1.232	1.218	230.0	227.3	6517	6189
3.20	5891	.653	.578	1.231	1.217	225.5	222.8	6581	6272
3.60	5769	.651	.580	1.231	1.216	220.8	218.1	6563	6263
4.00	5652	.650	.581	1.231	1.216	216.3	213.7	6509	6212
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	6054	.661	.573	1.235	1.220	232.4	229.6	6313	5930

PC= 1000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.228	.237	1.465	1.435	268.9	263.5	5535	3799
2.40	6048	1.215	.255	1.470	1.441	276.4	270.9	6242	4765
2.80	6004	1.206	.269	1.475	1.446	275.4	269.9	6517	5370
3.20	5891	1.201	.275	1.475	1.446	270.2	264.8	6581	5550
3.60	5769	1.201	.275	1.476	1.446	264.7	259.4	6563	5560
4.00	5652	1.201	.275	1.476	1.447	259.3	254.1	6509	5513
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.55	6054	1.211	.261	1.472	1.443	277.0	271.5	6398	5038

PC= 1000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.359	.190	1.549	1.505	284.3	276.2	5535	3368
2.40	6048	1.353	.208	1.561	1.516	293.4	285.1	6242	4331
2.80	6004	1.350	.210	1.560	1.516	291.1	282.9	6517	5054
3.20	5891	1.346	.216	1.562	1.517	286.0	277.9	6581	5294
3.60	5769	1.345	.217	1.562	1.518	280.2	272.3	6563	5315
4.00	5652	1.346	.217	1.563	1.519	274.5	266.8	6509	5268
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.52	6055	1.351	.210	1.562	1.517	293.9	285.6	6369	4571

PC= 1000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.434	.161	1.594	1.536	292.7	281.9	5535	3098
2.40	6048	1.433	.177	1.611	1.552	302.8	291.8	6242	4041
2.80	6004	1.431	.197	1.628	1.569	303.8	292.8	6517	4833
3.20	5891	1.427	.205	1.632	1.573	298.8	288.1	6581	5131
3.60	5769	1.427	.206	1.633	1.574	292.8	282.3	6563	5161
4.00	5652	1.427	.206	1.633	1.574	286.9	276.6	6509	5113
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.66	6027	1.432	.191	1.623	1.564	304.1	293.1	6447	4619

PC= 1000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.483	.147	1.630	1.556	299.2	285.7	5535	2909
2.40	6048	1.487	.163	1.650	1.576	310.2	296.4	6242	3828
2.80	6004	1.488	.183	1.671	1.598	311.9	298.2	6517	4659
3.20	5891	1.486	.192	1.677	1.604	307.2	293.7	6581	5008
3.60	5769	1.486	.193	1.679	1.605	301.1	287.9	6563	5047
4.00	5652	1.486	.193	1.679	1.605	295.0	282.0	6509	4998
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.70	6021	1.488	.179	1.667	1.594	312.1	298.3	6470	4503

TABLE III-13 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 1000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.519	.139	1.658	1.570	304.3	288.1	5535	2767
2.40	6048	1.526	.154	1.680	1.592	315.9	299.4	6242	3662
2.80	6004	1.532	.169	1.701	1.613	317.4	301.0	6517	4514
3.20	5891	1.531	.179	1.709	1.621	313.0	296.9	6581	4910
3.60	5769	1.530	.180	1.711	1.623	306.8	291.0	6563	4957
4.00	5652	1.531	.180	1.711	1.623	300.6	285.1	6509	4907
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.70	6021	1.531	.166	1.697	1.609	317.6	301.1	6469	4346

PC= 1000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.548	.128	1.677	1.574	307.8	288.9	5535	2650
2.40	6048	1.558	.144	1.702	1.599	320.0	300.7	6242	3526
2.80	6004	1.565	.164	1.729	1.626	322.7	303.5	6517	4392
3.20	5891	1.564	.175	1.739	1.636	318.4	299.6	6581	4832
3.60	5769	1.564	.177	1.741	1.638	312.2	293.8	6563	4886
4.00	5652	1.564	.177	1.741	1.638	305.9	287.8	6509	4835
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.76	6012	1.565	.162	1.727	1.624	322.8	303.5	6497	4317

PC= 1000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.571	.123	1.695	1.577	311.1	289.5	5535	2556
2.40	6048	1.584	.135	1.719	1.602	323.2	301.1	6242	3411
2.80	6004	1.594	.155	1.749	1.631	326.4	304.5	6517	4285
3.20	5891	1.593	.167	1.760	1.643	322.3	300.8	6581	4763
3.60	5769	1.593	.169	1.762	1.645	316.1	295.0	6563	4824
4.00	5652	1.593	.169	1.763	1.645	309.7	289.0	6509	4773
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	6009	1.593	.154	1.748	1.630	326.4	304.5	6506	4243

PC= 1000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.607	.113	1.721	1.574	315.9	288.9	5535	2407
2.40	6048	1.623	.126	1.749	1.602	328.9	301.3	6242	3231
2.80	6004	1.638	.145	1.783	1.636	332.7	305.3	6517	4105
3.20	5891	1.638	.159	1.796	1.649	329.0	302.1	6581	4651
3.60	5769	1.638	.161	1.799	1.652	322.6	296.2	6563	4726
4.00	5652	1.639	.161	1.799	1.652	316.1	290.3	6509	4671
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.80	6004	1.638	.145	1.783	1.636	332.7	305.3	6518	4107

PC= 1000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.635	.105	1.740	1.564	319.4	287.0	5535	2292
2.40	6048	1.653	.118	1.772	1.596	333.2	300.0	6242	3089
2.80	6004	1.671	.137	1.808	1.632	337.5	304.6	6517	3958
3.20	5891	1.673	.150	1.824	1.647	333.9	301.7	6581	4558
3.60	5769	1.673	.153	1.826	1.650	327.5	295.9	6563	4646
4.00	5652	1.674	.152	1.826	1.650	320.8	289.8	6509	4590
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.82	6000	1.672	.138	1.810	1.633	337.5	304.6	6525	3994

TABLE III-13 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 1000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.666	.098	1.764	1.543	323.7	283.3	5535	2161
2.40	6048	1.687	.111	1.798	1.577	338.0	296.6	6242	2927
2.80	6004	1.710	.126	1.836	1.616	342.8	301.6	6517	3779
3.20	5891	1.715	.142	1.857	1.637	340.1	299.7	6581	4442
3.60	5769	1.715	.145	1.860	1.640	333.6	294.0	6563	4550
4.00	5652	1.714	.145	1.859	1.638	326.6	287.8	6509	4494
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.86	5992	1.712	.129	1.840	1.620	342.8	301.8	6539	3883

PC= 1000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.701	.089	1.790	1.496	328.7	274.7	5535	2010
2.40	6048	1.727	.100	1.828	1.534	343.6	288.4	6242	2729
2.80	6004	1.756	.116	1.872	1.578	349.4	294.5	6517	3555
3.20	5891	1.763	.133	1.897	1.603	347.3	293.5	6581	4291
3.60	5769	1.763	.137	1.900	1.606	340.8	288.1	6563	4433
4.00	5652	1.764	.137	1.900	1.607	333.9	282.3	6509	4370
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.90	5983	1.759	.120	1.880	1.586	349.6	294.9	6552	3740

PC= 1000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.746	.079	1.824	1.383	334.9	254.0	5535	1821
2.40	6048	1.777	.088	1.865	1.425	350.7	267.8	6242	2477
2.80	6004	1.813	.104	1.917	1.476	357.8	275.5	6517	3256
3.20	5891	1.827	.121	1.948	1.507	356.7	276.0	6581	4064
3.60	5769	1.828	.126	1.954	1.513	350.4	271.3	6563	4271
4.00	5652	1.828	.125	1.954	1.513	343.3	265.8	6509	4198
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.95	5970	1.821	.110	1.931	1.490	358.3	276.5	6566	3555

PC= 1000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.773	.072	1.846	1.258	338.8	230.9	5535	1704
2.40	6048	1.809	.081	1.890	1.302	355.3	244.8	6242	2315
2.80	6004	1.850	.095	1.945	1.357	363.1	253.4	6517	3056
3.20	5891	1.869	.113	1.982	1.394	363.0	255.3	6581	3888
3.60	5769	1.869	.119	1.988	1.400	356.6	251.2	6563	4160
4.00	5652	1.870	.118	1.988	1.400	349.3	246.0	6509	4078
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.99	5958	1.862	.103	1.965	1.378	364.0	255.1	6575	3449

PC= 1000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5905	1.793	.068	1.862	1.127	341.7	206.9	5535	1624
2.40	6048	1.831	.077	1.908	1.173	358.7	220.6	6242	2197
2.80	6004	1.877	.089	1.966	1.231	366.8	229.7	6517	2910
3.20	5891	1.900	.107	2.007	1.272	367.5	232.9	6581	3742
3.60	5769	1.901	.114	2.015	1.280	361.3	229.5	6563	4073
4.00	5652	1.900	.113	2.014	1.279	353.8	224.7	6509	3985
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
3.04	5938	1.893	.101	1.994	1.259	368.0	232.4	6565	3464

TABLE III-14

LOX/C_3H_8, $P_c = 2,000$

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 2000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	.680	.563	1.243	1.236	228.4	227.1	5579	5068
2.40	6073	.666	.571	1.236	1.229	233.4	232.0	6359	5930
2.80	6047	.657	.576	1.233	1.226	231.8	230.4	6688	6332
3.20	5939	.652	.580	1.232	1.225	227.4	226.1	6769	6441
3.60	5816	.652	.579	1.232	1.224	222.7	221.3	6752	6431
4.00	5697	.653	.579	1.231	1.224	218.1	216.8	6693	6375
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.50	6084	.663	.572	1.235	1.228	233.6	232.3	6485	6075

PC= 2000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.229	.236	1.465	1.450	269.2	266.5	5579	3797
2.40	6073	1.217	.252	1.469	1.454	277.3	274.6	6359	4772
2.80	6047	1.208	.267	1.475	1.460	277.3	274.5	6688	5444
3.20	5939	1.204	.272	1.477	1.462	272.6	269.9	6769	5663
3.60	5816	1.203	.274	1.477	1.462	267.0	264.4	6752	5681
4.00	5697	1.203	.274	1.477	1.462	261.6	259.0	6693	5629
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.60	6084	1.212	.260	1.472	1.457	278.3	275.6	6577	5140

PC= 2000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.359	.189	1.548	1.526	284.6	280.5	5579	3364
2.40	6073	1.353	.205	1.559	1.537	294.3	290.1	6359	4321
2.80	6047	1.352	.207	1.559	1.537	293.0	288.9	6688	5094
3.20	5939	1.348	.214	1.562	1.540	288.3	284.3	6769	5391
3.60	5816	1.347	.215	1.563	1.541	282.5	278.5	6752	5422
4.00	5697	1.347	.215	1.563	1.541	276.8	272.9	6693	5370
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.55	6085	1.352	.208	1.560	1.538	295.1	290.9	6540	4641

PC= 2000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.434	.160	1.594	1.565	292.9	287.5	5579	3095
2.40	6073	1.433	.175	1.608	1.579	303.6	298.1	6359	4025
2.80	6047	1.432	.194	1.625	1.596	305.5	300.0	6688	4852
3.20	5939	1.428	.203	1.631	1.602	301.1	295.7	6769	5217
3.60	5816	1.427	.205	1.632	1.603	295.1	289.8	6752	5260
4.00	5697	1.428	.205	1.632	1.603	289.1	283.9	6693	5207
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.72	6059	1.432	.191	1.623	1.594	305.7	300.1	6643	4725

PC= 2000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.483	.146	1.629	1.593	299.4	292.7	5579	2906
2.40	6073	1.486	.161	1.647	1.610	311.0	304.0	6359	3810
2.80	6047	1.488	.180	1.668	1.631	313.5	306.6	6688	4663
3.20	5939	1.485	.190	1.676	1.639	309.3	302.5	6769	5087
3.60	5816	1.485	.192	1.677	1.641	303.3	296.6	6752	5139
4.00	5697	1.485	.192	1.677	1.641	297.1	290.6	6693	5085
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.75	6054	1.488	.178	1.666	1.629	313.6	306.7	6662	4584

TABLE III-14 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.2,1960

PC= 2000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.519	.138	1.657	1.613	304.6	290.5	5579	2764
2.40	6073	1.525	.152	1.677	1.633	316.6	308.3	6359	3643
2.80	6047	1.531	.166	1.697	1.653	319.0	310.7	6688	4508
3.20	5939	1.530	.177	1.707	1.663	315.1	307.0	6769	4982
3.60	5816	1.530	.179	1.709	1.665	309.0	301.0	6752	5044
4.00	5697	1.530	.179	1.709	1.665	302.7	294.9	6693	4989
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.75	6055	1.531	.164	1.695	1.651	319.0	310.8	6661	4422

PC= 2000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.548	.128	1.676	1.625	308.0	298.6	5579	2647
2.40	6073	1.556	.142	1.698	1.647	320.7	310.9	6359	3506
2.80	6047	1.564	.161	1.725	1.674	324.3	314.6	6688	4377
3.20	5939	1.563	.174	1.736	1.685	320.6	311.1	6769	4898
3.60	5816	1.564	.176	1.739	1.688	314.5	305.2	6752	4968
4.00	5697	1.564	.175	1.740	1.688	308.1	299.0	6693	4911
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.80	6047	1.564	.161	1.725	1.673	324.3	314.6	6687	4372

PC= 2000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.571	.123	1.694	1.635	311.3	300.5	5579	2553
2.40	6073	1.582	.133	1.715	1.657	323.9	312.8	6359	3392
2.80	6047	1.592	.152	1.744	1.685	327.9	316.8	6688	4263
3.20	5939	1.592	.166	1.757	1.699	324.5	313.6	6769	4823
3.60	5816	1.593	.168	1.761	1.702	318.4	307.8	6752	4902
4.00	5697	1.593	.168	1.761	1.702	311.9	301.5	6693	4844
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.82	6044	1.593	.153	1.745	1.686	327.9	316.8	6696	4291

PC= 2000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.607	.113	1.720	1.646	316.1	302.6	5579	2405
2.40	6073	1.621	.124	1.745	1.672	329.5	315.6	6359	3212
2.80	6047	1.636	.142	1.778	1.704	334.2	320.4	6688	4075
3.20	5939	1.637	.157	1.794	1.721	331.2	317.7	6769	4698
3.60	5816	1.637	.160	1.797	1.723	324.9	311.6	6752	4798
4.00	5697	1.638	.159	1.797	1.724	318.3	305.3	6693	4737
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.85	6039	1.636	.144	1.780	1.707	334.2	320.4	6710	4157

PC= 2000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.634	.105	1.739	1.651	319.6	303.4	5579	2290
2.40	6073	1.651	.117	1.768	1.679	333.7	317.1	6359	3071
2.80	6047	1.669	.134	1.803	1.715	338.9	322.3	6688	3923
3.20	5939	1.673	.148	1.821	1.733	336.1	319.8	6769	4595
3.60	5816	1.673	.151	1.824	1.736	329.8	313.8	6752	4713
4.00	5697	1.673	.151	1.824	1.736	323.0	307.4	6693	4650
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.86	6036	1.670	.136	1.807	1.718	339.0	322.4	6716	4037

TABLE III-14 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 26A 03 DEC 60

LO2(NBP)/C3H8(NBP) NOV.8,1960

PC= 2000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.665	.097	1.763	1.652	323.9	303.7	5579	2159
2.40	6073	1.684	.109	1.793	1.683	338.5	317.7	6359	2909
2.80	6047	1.707	.124	1.830	1.720	344.0	323.3	6688	3741
3.20	5939	1.712	.140	1.852	1.742	342.0	321.6	6769	4471
3.60	5816	1.714	.144	1.857	1.747	335.8	315.9	6752	4612
4.00	5697	1.713	.144	1.856	1.746	328.8	309.2	6693	4548
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.89	6030	1.709	.127	1.837	1.726	344.2	323.6	6728	3915

PC= 2000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.700	.089	1.789	1.642	328.8	301.8	5579	2008
2.40	6073	1.724	.099	1.823	1.676	344.1	316.4	6359	2713
2.80	6047	1.751	.114	1.866	1.719	350.7	323.1	6688	3516
3.20	5939	1.762	.131	1.893	1.746	349.4	322.3	6769	4298
3.60	5816	1.762	.136	1.898	1.751	343.1	316.5	6752	4487
4.00	5697	1.761	.135	1.897	1.750	335.9	309.9	6693	4416
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.93	6020	1.757	.120	1.877	1.730	351.2	323.7	6743	3780

PC= 2000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.745	.079	1.823	1.603	335.1	294.5	5579	1821
2.40	6073	1.773	.087	1.860	1.640	351.2	309.5	6359	2462
2.80	6047	1.808	.101	1.909	1.688	358.8	317.4	6688	3218
3.20	5939	1.824	.118	1.943	1.722	358.7	318.0	6769	4043
3.60	5816	1.826	.125	1.950	1.730	352.6	312.8	6752	4315
4.00	5697	1.826	.124	1.950	1.730	345.4	306.3	6693	4230
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.99	6005	1.818	.109	1.927	1.707	359.7	318.6	6758	3604

PC= 2000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.773	.072	1.845	1.551	339.1	285.0	5579	1707
2.40	6073	1.804	.080	1.884	1.590	355.7	300.2	6359	2301
2.80	6047	1.844	.093	1.937	1.643	364.1	308.8	6688	3019
3.20	5939	1.866	.110	1.976	1.682	364.8	310.6	6769	3847
3.60	5816	1.867	.118	1.985	1.691	358.8	305.7	6752	4196
4.00	5697	1.867	.116	1.984	1.690	351.4	299.3	6693	4100
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
3.05	5982	1.860	.105	1.965	1.671	365.3	310.7	6750	3585

PC= 2000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000000071

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
2.00	5912	1.792	.068	1.861	1.493	342.0	274.5	5579	1634
2.40	6073	1.826	.076	1.902	1.535	359.1	289.7	6359	2184
2.80	6047	1.870	.087	1.956	1.589	367.8	298.7	6688	2875
3.20	5939	1.896	.104	2.000	1.632	369.2	301.4	6769	3694
3.60	5816	1.898	.113	2.011	1.644	363.6	297.2	6752	4102
4.00	5697	1.898	.111	2.009	1.642	355.8	290.8	6693	3999
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
3.08	5972	1.891	.100	1.990	1.623	369.5	301.3	6755	3494

TABLE III-15

LOX/RP-1, P_c = 300

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 300 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	.681	.562	1.242	1.193	221.8	213.0	5419	4949
2.00	5857	.669	.569	1.238	1.189	225.3	216.4	5836	5431
2.20	5891	.661	.574	1.235	1.186	226.2	217.2	6091	5744
2.40	5873	.656	.577	1.233	1.184	225.0	216.1	6229	5918
2.60	5826	.654	.578	1.232	1.183	223.0	214.2	6298	6005
2.80	5766	.653	.579	1.231	1.182	220.7	211.9	6327	6044
3.00	5705	.652	.580	1.231	1.182	218.4	209.7	6332	6056
3.20	5644	.652	.579	1.231	1.182	216.0	207.4	6323	6051
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.18	5890	.662	.574	1.235	1.186	226.2	217.2	6075	5724

PC= 300 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.229	.235	1.464	1.366	261.4	243.9	5419	3713
2.00	5857	1.221	.247	1.468	1.370	267.2	249.4	5836	4295
2.20	5891	1.212	.260	1.472	1.374	269.5	251.6	6091	4787
2.40	5873	1.206	.269	1.474	1.376	269.1	251.3	6229	5122
2.60	5826	1.203	.273	1.476	1.378	267.3	249.5	6298	5297
2.80	5766	1.201	.275	1.477	1.379	264.7	247.1	6327	5377
3.00	5705	1.201	.276	1.477	1.379	261.9	244.6	6332	5409
3.20	5644	1.200	.277	1.477	1.379	259.2	242.0	6323	5414
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.27	5890	1.210	.263	1.473	1.375	269.7	251.7	6155	4929

PC= 300 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.360	.187	1.547	1.400	276.2	250.0	5419	3277
2.00	5857	1.356	.198	1.555	1.408	283.0	256.3	5836	3846
2.20	5891	1.352	.211	1.563	1.416	286.3	259.4	6091	4379
2.40	5873	1.350	.208	1.559	1.412	284.5	257.7	6229	4796
2.60	5826	1.348	.214	1.562	1.415	282.8	256.2	6298	5031
2.80	5766	1.347	.217	1.564	1.417	280.2	253.9	6327	5138
3.00	5705	1.346	.218	1.564	1.417	277.4	251.3	6332	5182
3.20	5644	1.346	.219	1.565	1.418	274.5	248.7	6323	5192
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.23	5891	1.352	.212	1.564	1.417	286.3	259.4	6119	4448

PC= 300 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.434	.161	1.595	1.399	284.8	249.8	5419	3005
2.00	5857	1.434	.171	1.605	1.409	292.2	256.6	5836	3556
2.20	5891	1.434	.182	1.616	1.420	296.0	260.1	6091	4098
2.40	5873	1.431	.194	1.626	1.430	296.8	261.0	6229	4562
2.60	5826	1.429	.202	1.631	1.435	295.4	259.9	6298	4851
2.80	5766	1.428	.206	1.634	1.438	292.8	257.7	6327	4984
3.00	5705	1.428	.207	1.635	1.439	289.9	255.2	6332	5037
3.20	5644	1.428	.208	1.635	1.439	286.9	252.5	6323	5053
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.37	5877	1.432	.193	1.625	1.429	296.8	261.0	6214	4510

TABLE III-15 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 300 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.484	.145	1.629	1.384	290.8	247.1	5419	2816
2.00	5857	1.486	.155	1.641	1.396	298.8	254.2	5836	3347
2.20	5891	1.489	.167	1.656	1.411	303.2	258.4	6091	3886
2.40	5873	1.489	.179	1.668	1.423	304.4	259.7	6229	4377
2.60	5826	1.487	.187	1.674	1.429	303.2	258.8	6298	4711
2.80	5766	1.487	.191	1.678	1.433	300.7	256.8	6327	4868
3.00	5705	1.486	.193	1.679	1.434	297.7	254.3	6332	4931
3.20	5644	1.486	.194	1.680	1.435	294.7	251.7	6323	4950
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.40	5873	1.489	.179	1.668	1.423	304.4	259.7	6229	4377

PC= 300 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.520	.135	1.655	1.361	295.4	242.9	5419	2672
2.00	5857	1.525	.144	1.669	1.375	303.8	250.3	5836	3186
2.20	5891	1.530	.155	1.685	1.391	308.6	254.8	6091	3718
2.40	5873	1.531	.168	1.699	1.405	310.2	256.6	6229	4224
2.60	5826	1.531	.176	1.707	1.413	309.2	255.9	6298	4595
2.80	5766	1.531	.181	1.712	1.418	306.8	254.1	6327	4777
3.00	5705	1.530	.183	1.713	1.419	303.8	251.7	6332	4847
3.20	5644	1.530	.184	1.714	1.420	300.7	249.1	6323	4869
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.42	5869	1.531	.169	1.700	1.407	310.2	256.6	6240	4272

PC= 300 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.548	.126	1.675	1.332	299.0	237.7	5419	2557
2.00	5857	1.555	.135	1.690	1.347	307.7	245.3	5836	3057
2.20	5891	1.562	.147	1.708	1.366	312.8	250.1	6091	3579
2.40	5873	1.565	.158	1.724	1.381	314.7	252.1	6229	4094
2.60	5826	1.566	.168	1.734	1.391	314.0	251.9	6298	4495
2.80	5766	1.566	.174	1.739	1.396	311.7	250.3	6327	4699
3.00	5705	1.565	.176	1.741	1.398	308.8	247.9	6332	4778
3.20	5644	1.565	.177	1.742	1.399	305.6	245.4	6323	4803
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.45	5864	1.566	.161	1.727	1.384	314.8	252.3	6251	4198

PC= 300 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.571	.121	1.692	1.300	302.0	232.1	5419	2464
2.00	5857	1.579	.130	1.708	1.316	311.0	239.7	5836	2950
2.20	5891	1.588	.139	1.727	1.335	316.2	244.4	6091	3462
2.40	5873	1.593	.151	1.745	1.353	318.5	246.9	6229	3980
2.60	5826	1.594	.162	1.756	1.364	318.0	247.1	6298	4407
2.80	5766	1.594	.168	1.762	1.370	315.8	245.6	6327	4633
3.00	5705	1.594	.170	1.764	1.372	312.9	243.4	6332	4721
3.20	5644	1.594	.171	1.765	1.373	309.7	240.9	6323	4748
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.47	5860	1.594	.155	1.749	1.357	318.6	247.3	6260	4136

TABLE III-15 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 300 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.607	.111	1.718	1.228	306.6	219.2	5419	2317
2.00	5857	1.617	.118	1.735	1.245	315.9	226.7	5836	2780
2.20	5891	1.629	.128	1.757	1.267	321.7	232.0	6091	3277
2.40	5873	1.637	.140	1.778	1.288	324.5	235.1	6229	3791
2.60	5826	1.639	.152	1.792	1.302	324.5	235.8	6298	4252
2.80	5766	1.640	.159	1.799	1.309	322.5	234.7	6327	4521
3.00	5705	1.640	.162	1.802	1.312	319.5	232.6	6332	4625
3.20	5644	1.640	.163	1.803	1.313	316.3	230.4	6323	4658
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.50	5853	1.639	.146	1.785	1.296	324.9	235.7	6272	4024

PC= 300 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.633	.103	1.736	1.148	310.0	205.0	5419	2204
2.00	5857	1.645	.111	1.756	1.168	319.7	212.7	5836	2649
2.20	5891	1.659	.121	1.780	1.192	325.9	218.3	6091	3132
2.40	5873	1.670	.132	1.802	1.215	329.0	221.7	6229	3641
2.60	5826	1.675	.143	1.818	1.230	329.2	222.7	6298	4122
2.80	5766	1.676	.151	1.826	1.238	327.3	222.0	6327	4427
3.00	5705	1.676	.154	1.829	1.242	324.4	220.2	6332	4548
3.20	5644	1.676	.155	1.831	1.243	321.2	218.1	6323	4586
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.51	5848	1.673	.139	1.812	1.224	329.4	222.5	6273	3933

PC= 300 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.663	.096	1.759	1.024	314.0	182.9	5419	2078
2.00	5857	1.677	.103	1.780	1.045	324.1	190.3	5836	2501
2.20	5891	1.694	.112	1.806	1.071	330.7	196.2	6091	2963
2.40	5873	1.708	.121	1.830	1.095	334.0	199.9	6229	3460
2.60	5826	1.715	.134	1.848	1.114	334.8	201.7	6298	3957
2.80	5766	1.716	.143	1.859	1.124	333.2	201.5	6327	4310
3.00	5705	1.717	.146	1.863	1.128	330.4	200.1	6332	4456
3.20	5644	1.717	.147	1.865	1.130	327.1	198.2	6323	4500
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.56	5835	1.714	.132	1.846	1.111	334.8	201.5	6289	3880

PC= 300 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.698	.087	1.786	.806	318.8	143.9	5419	1929
2.00	5857	1.715	.093	1.808	.828	329.2	150.8	5836	2325
2.20	5891	1.735	.102	1.837	.857	336.3	156.9	6091	2760
2.40	5873	1.752	.112	1.864	.885	340.4	161.5	6229	3239
2.60	5826	1.763	.123	1.886	.906	341.5	164.1	6298	3740
2.80	5766	1.766	.133	1.899	.919	340.3	164.8	6327	4151
3.00	5705	1.766	.138	1.904	.924	337.7	163.9	6332	4338
3.20	5644	1.767	.139	1.906	.927	334.4	162.5	6323	4394
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.60	5826	1.763	.123	1.886	.906	341.5	164.1	6298	3738

TABLE III-15 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 300 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.742	.077	1.819	.349	324.7	62.4	5419	1743
2.00	5857	1.762	.082	1.844	.374	335.7	68.2	5836	2103
2.20	5891	1.786	.089	1.875	.405	343.3	74.2	6091	2501
2.40	5873	1.808	.099	1.907	.437	348.1	79.8	6229	2947
2.60	5826	1.824	.109	1.933	.464	350.1	84.0	6298	3436
2.80	5766	1.830	.120	1.950	.481	349.6	86.2	6327	3906
3.00	5705	1.831	.127	1.958	.488	347.1	86.5	6332	4168
3.20	5644	1.832	.128	1.960	.491	343.9	86.1	6323	4247
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.66	5810	1.827	.112	1.939	.469	350.2	84.8	6310	3573

PC= 300 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.769	.072	1.841	.000	328.7	.0	5419	1627
2.00	5857	1.791	.076	1.867	.000	339.9	.0	5836	1962
2.20	5891	1.818	.082	1.900	.000	347.8	.0	6091	2335
2.40	5873	1.844	.090	1.934	.000	353.0	.0	6229	2757
2.60	5826	1.863	.101	1.963	.004	355.6	.7	6298	3230
2.80	5766	1.872	.112	1.984	.025	355.6	4.4	6327	3717
3.00	5705	1.874	.119	1.992	.033	353.3	5.9	6332	4044
3.20	5644	1.874	.121	1.996	.036	350.1	6.4	6323	4146
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.70	5796	1.868	.107	1.975	.016	355.9	2.9	6316	3496

PC= 300 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5743	1.787	.080	1.867	.000	333.3	.0	5419	1552
2.00	5857	1.811	.080	1.891	.000	344.3	.0	5836	1867
2.20	5891	1.839	.082	1.921	.000	351.8	.0	6091	2221
2.40	5873	1.868	.087	1.955	.000	357.0	.0	6229	2620
2.60	5826	1.891	.095	1.986	.000	359.6	.0	6298	3076
2.80	5766	1.903	.105	2.008	.000	359.8	.0	6327	3565
3.00	5705	1.905	.114	2.019	.000	358.0	.0	6332	3942
3.20	5644	1.906	.117	2.022	.000	354.8	.0	6323	4068
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.73	5788	1.899	.102	2.001	.000	360.0	.0	6319	3397

TABLE III-16

LOX/RP-1, P_c = 400

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 400 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	.681	.561	1.243	1.206	222.0	215.5	5445	4963
2.00	5867	.670	.568	1.238	1.202	225.8	219.1	5880	5461
2.20	5906	.662	.573	1.235	1.198	226.8	220.0	6150	5790
2.40	5892	.656	.577	1.233	1.196	225.8	219.1	6298	5977
2.60	5845	.654	.578	1.232	1.196	223.9	217.2	6372	6071
2.80	5788	.653	.579	1.232	1.195	221.6	215.0	6404	6113
3.00	5727	.652	.579	1.232	1.195	219.2	212.7	6410	6126
3.20	5665	.652	.580	1.231	1.195	216.9	210.4	6401	6121
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.20	5906	.662	.573	1.235	1.198	226.8	220.0	6150	5790

PC= 400 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.230	.235	1.464	1.391	261.6	248.5	5445	3711
2.00	5867	1.221	.246	1.467	1.394	267.6	254.2	5880	4294
2.20	5906	1.213	.258	1.471	1.398	270.1	256.6	6150	4798
2.40	5892	1.207	.267	1.474	1.401	270.0	256.5	6298	5150
2.60	5845	1.203	.273	1.476	1.403	268.2	254.8	6372	5339
2.80	5788	1.202	.275	1.477	1.403	265.7	252.4	6404	5425
3.00	5727	1.201	.276	1.477	1.403	262.9	249.8	6410	5460
3.20	5665	1.201	.276	1.477	1.404	260.1	247.2	6401	5466
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.29	5906	1.210	.263	1.473	1.399	270.4	256.9	6231	4974

PC= 400 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.360	.187	1.547	1.437	276.4	256.7	5445	3274
2.00	5867	1.356	.197	1.554	1.444	283.4	263.3	5880	3842
2.20	5906	1.353	.210	1.563	1.452	286.9	266.6	6150	4378
2.40	5892	1.351	.207	1.558	1.448	285.3	265.1	6298	4810
2.60	5845	1.348	.213	1.562	1.451	283.7	263.7	6372	5062
2.80	5788	1.347	.216	1.563	1.453	281.2	261.4	6404	5179
3.00	5727	1.346	.217	1.564	1.453	278.4	258.7	6410	5227
3.20	5665	1.346	.218	1.564	1.454	275.5	256.1	6401	5239
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.24	5907	1.352	.210	1.563	1.453	287.0	266.7	6188	4468

PC= 400 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.434	.161	1.595	1.448	285.0	258.7	5445	3003
2.00	5867	1.434	.170	1.604	1.457	292.6	265.8	5880	3549
2.20	5906	1.434	.181	1.615	1.468	296.5	269.5	6150	4092
2.40	5892	1.432	.193	1.625	1.478	297.5	270.6	6298	4567
2.60	5845	1.430	.201	1.631	1.484	296.3	269.6	6372	4875
2.80	5788	1.429	.205	1.633	1.486	293.8	267.4	6404	5020
3.00	5727	1.428	.206	1.634	1.487	290.9	264.8	6410	5078
3.20	5665	1.428	.207	1.635	1.488	287.9	262.1	6401	5095
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.39	5893	1.432	.192	1.624	1.477	297.5	270.6	6293	4548

TABLE III-16 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57.

PC= 400 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.483	.145	1.629	1.445	291.0	258.2	5445	2813
2.00	5867	1.486	.154	1.640	1.456	299.1	265.6	5880	3340
2.20	5906	1.489	.166	1.654	1.471	303.7	270.0	6150	3877
2.40	5892	1.489	.178	1.666	1.483	305.2	271.5	6298	4375
2.60	5845	1.487	.186	1.674	1.490	304.1	270.7	6372	4730
2.80	5788	1.486	.191	1.677	1.493	301.7	268.6	6404	4902
3.00	5727	1.486	.192	1.678	1.495	298.8	266.1	6410	4969
3.20	5665	1.486	.193	1.679	1.495	295.7	263.3	6401	4990
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.42	5889	1.489	.178	1.667	1.483	305.2	271.6	6306	4408

PC= 400 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.520	.134	1.654	1.434	295.6	256.2	5445	2669
2.00	5867	1.524	.143	1.667	1.447	304.1	263.9	5880	3179
2.20	5906	1.529	.154	1.683	1.463	309.1	268.6	6150	3707
2.40	5892	1.531	.166	1.698	1.477	310.9	270.6	6298	4218
2.60	5845	1.531	.175	1.707	1.486	310.1	270.0	6372	4609
2.80	5788	1.531	.180	1.711	1.490	307.8	268.1	6404	4805
3.00	5727	1.530	.182	1.712	1.492	304.8	265.6	6410	4884
3.20	5665	1.530	.183	1.713	1.493	301.7	262.9	6401	4908
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.44	5886	1.532	.168	1.700	1.479	311.0	270.7	6317	4297

PC= 400 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.548	.126	1.674	1.417	299.1	253.2	5445	2555
2.00	5867	1.554	.135	1.689	1.432	308.0	261.1	5880	3049
2.20	5906	1.561	.145	1.707	1.449	313.3	266.1	6150	3567
2.40	5892	1.565	.157	1.722	1.465	315.4	268.3	6298	4084
2.60	5845	1.566	.167	1.733	1.476	314.9	268.2	6372	4503
2.80	5788	1.565	.173	1.738	1.481	312.7	266.4	6404	4725
3.00	5727	1.565	.175	1.740	1.483	309.8	264.0	6410	4812
3.20	5665	1.565	.176	1.741	1.484	306.6	261.3	6401	4840
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	5881	1.566	.160	1.726	1.469	315.5	268.5	6329	4223

PC= 400 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.571	.121	1.691	1.398	302.2	249.7	5445	2462
2.00	5867	1.578	.129	1.707	1.413	311.3	257.7	5880	2944
2.20	5906	1.588	.137	1.725	1.431	316.7	262.7	6150	3450
2.40	5892	1.593	.150	1.743	1.449	319.1	265.3	6298	3968
2.60	5845	1.595	.161	1.755	1.461	318.9	265.5	6372	4410
2.80	5788	1.594	.167	1.761	1.467	316.8	263.9	6404	4656
3.00	5727	1.594	.169	1.763	1.469	313.9	261.6	6410	4753
3.20	5665	1.594	.171	1.764	1.471	310.7	259.0	6401	4783
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.48	5876	1.594	.154	1.748	1.455	319.4	265.7	6338	4161

TABLE III-16 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 400 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.606	.111	1.717	1.350	306.8	241.1	5445	2314
2.00	5867	1.616	.117	1.733	1.366	316.1	249.1	5880	2774
2.20	5906	1.628	.127	1.755	1.387	322.2	254.7	6150	3264
2.40	5892	1.636	.139	1.775	1.408	325.1	257.8	6298	3778
2.60	5845	1.640	.151	1.791	1.423	325.3	258.6	6372	4249
2.80	5788	1.639	.158	1.798	1.430	323.4	257.3	6404	4540
3.00	5727	1.639	.161	1.801	1.433	320.5	255.1	6410	4655
3.20	5665	1.640	.162	1.802	1.435	317.3	252.6	6401	4690
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.52	5865	1.639	.147	1.785	1.418	325.5	258.5	6346	4076

PC= 400 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.633	.103	1.736	1.295	310.1	231.4	5445	2201
2.00	5867	1.644	.110	1.754	1.313	319.9	239.5	5880	2644
2.20	5906	1.658	.120	1.778	1.337	326.3	245.4	6150	3120
2.40	5892	1.669	.131	1.800	1.359	329.7	248.9	6298	3626
2.60	5845	1.675	.142	1.816	1.375	330.0	249.9	6372	4113
2.80	5788	1.675	.150	1.825	1.384	328.3	248.9	6404	4442
3.00	5727	1.675	.153	1.828	1.387	325.4	247.0	6410	4575
3.20	5665	1.676	.154	1.830	1.389	322.2	244.6	6401	4616
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.53	5862	1.673	.138	1.811	1.370	330.1	249.7	6351	3962

PC= 400 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.663	.096	1.759	1.207	314.2	215.7	5445	2075
2.00	5867	1.676	.102	1.778	1.227	324.3	223.8	5880	2496
2.20	5906	1.692	.111	1.804	1.252	331.1	229.9	6150	2952
2.40	5892	1.706	.122	1.828	1.277	334.8	233.9	6298	3445
2.60	5845	1.715	.132	1.847	1.296	335.5	235.4	6372	3943
2.80	5788	1.716	.142	1.857	1.306	334.2	235.0	6404	4319
3.00	5727	1.716	.146	1.862	1.311	331.4	233.3	6410	4480
3.20	5665	1.717	.147	1.864	1.313	328.2	231.1	6401	4530
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.57	5853	1.714	.131	1.844	1.293	335.6	235.3	6363	3869

PC= 400 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.698	.087	1.785	1.050	319.0	187.7	5445	1925
2.00	5867	1.713	.093	1.806	1.071	329.4	195.4	5880	2318
2.20	5906	1.733	.101	1.834	1.099	336.6	201.7	6150	2749
2.40	5892	1.750	.111	1.861	1.126	340.9	206.3	6298	3222
2.60	5845	1.762	.121	1.884	1.149	342.2	208.7	6372	3722
2.80	5788	1.765	.132	1.897	1.162	341.3	209.1	6404	4151
3.00	5727	1.766	.137	1.903	1.168	338.7	207.9	6410	4357
3.20	5665	1.766	.139	1.905	1.170	335.4	206.1	6401	4420
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.62	5840	1.763	.122	1.885	1.150	342.2	208.8	6376	3762

TABLE III-16 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 400 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.742	.077	1.819	.716	324.9	128.0	5445	1737
2.00	5867	1.760	.082	1.842	.740	335.9	134.9	5880	2097
2.20	5906	1.783	.088	1.872	.770	343.6	141.3	6150	2491
2.40	5892	1.806	.097	1.903	.801	348.5	146.7	6298	2932
2.60	5845	1.822	.108	1.931	.828	350.8	150.5	6372	3416
2.80	5788	1.829	.119	1.948	.846	350.4	152.1	6404	3893
3.00	5727	1.830	.126	1.956	.853	348.1	151.9	6410	4181
3.20	5665	1.831	.128	1.959	.856	344.9	150.8	6401	4269
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.67	5824	1.826	.112	1.938	.836	350.9	151.4	6389	3596

PC= 400 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.769	.072	1.841	.372	329.0	66.4	5445	1620
2.00	5867	1.789	.076	1.865	.395	340.1	72.1	5880	1956
2.20	5906	1.815	.081	1.896	.427	348.1	78.3	6150	2325
2.40	5892	1.841	.089	1.930	.460	353.4	84.3	6298	2742
2.60	5845	1.861	.099	1.960	.491	356.2	89.2	6372	3209
2.80	5788	1.870	.111	1.981	.511	356.4	92.0	6404	3697
3.00	5727	1.872	.118	1.990	.521	354.3	92.7	6410	4050
3.20	5665	1.873	.121	1.994	.525	351.2	92.4	6401	4165
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.72	5811	1.867	.106	1.974	.504	356.6	91.1	6394	3516

PC= 400 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5748	1.787	.080	1.868	.031	333.7	5.5	5445	1544
2.00	5867	1.809	.080	1.889	.052	344.5	9.5	5880	1862
2.20	5906	1.837	.082	1.919	.082	352.3	15.1	6150	2212
2.40	5892	1.865	.087	1.952	.115	357.4	21.0	6298	2606
2.60	5845	1.889	.094	1.982	.145	360.2	26.4	6372	3054
2.80	5788	1.901	.104	2.005	.168	360.6	30.1	6404	3542
3.00	5727	1.904	.113	2.016	.179	358.9	31.9	6410	3942
3.20	5665	1.905	.116	2.021	.184	355.8	32.3	6401	4084
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.74	5805	1.898	.101	1.999	.162	360.7	29.3	6397	3410

TABLE III-17

LOX/RP-1, P_c = 500

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 500 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	.682	.561	1.243	1.214	222.1	216.9	5460	4970
2.00	5872	.671	.568	1.239	1.210	226.2	220.8	5911	5480
2.20	5914	.661	.574	1.235	1.205	227.0	221.6	6187	5817
2.40	5901	.656	.576	1.233	1.203	226.2	220.8	6346	6016
2.60	5856	.655	.577	1.232	1.203	224.3	219.0	6426	6117
2.80	5804	.653	.579	1.232	1.203	222.3	217.0	6464	6167
3.00	5743	.652	.579	1.232	1.202	219.9	214.6	6472	6181
3.20	5681	.652	.580	1.232	1.202	217.5	212.3	6463	6177
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.20	5914	.661	.574	1.235	1.205	227.0	221.6	6186	5816

PC= 500 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.230	.234	1.464	1.405	261.6	251.1	5460	3709
2.00	5872	1.222	.245	1.467	1.408	267.8	257.1	5911	4291
2.20	5914	1.213	.257	1.470	1.411	270.3	259.5	6187	4797
2.40	5901	1.207	.267	1.474	1.415	270.3	259.5	6346	5163
2.60	5856	1.204	.272	1.476	1.417	268.6	257.9	6426	5366
2.80	5804	1.202	.274	1.476	1.418	266.4	255.8	6464	5464
3.00	5743	1.201	.275	1.477	1.418	263.6	253.1	6472	5501
3.20	5681	1.201	.276	1.477	1.418	260.8	250.5	6463	5509
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.30	5911	1.210	.263	1.472	1.413	270.5	259.7	6278	5005

PC= 500 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.360	.186	1.547	1.459	276.4	260.7	5460	3270
2.00	5872	1.357	.197	1.554	1.465	283.6	267.5	5911	3836
2.20	5914	1.352	.210	1.562	1.474	287.1	270.9	6187	4373
2.40	5901	1.351	.206	1.557	1.469	285.6	269.4	6346	4813
2.60	5856	1.348	.213	1.561	1.473	284.2	268.1	6426	5082
2.80	5804	1.347	.216	1.563	1.475	281.9	266.0	6464	5212
3.00	5743	1.346	.217	1.564	1.475	279.1	263.4	6472	5263
3.20	5681	1.346	.218	1.564	1.476	276.2	260.6	6463	5277
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.24	5916	1.351	.210	1.562	1.474	287.2	271.0	6228	4470

PC= 500 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.434	.160	1.595	1.477	285.0	263.9	5460	3000
2.00	5872	1.434	.170	1.604	1.486	292.7	271.2	5911	3544
2.20	5914	1.433	.179	1.613	1.495	296.5	274.9	6187	4085
2.40	5901	1.432	.191	1.623	1.506	297.7	276.2	6346	4564
2.60	5856	1.430	.200	1.630	1.512	296.7	275.3	6426	4889
2.80	5804	1.428	.204	1.633	1.515	294.6	273.4	6464	5048
3.00	5743	1.428	.206	1.634	1.517	291.7	270.7	6472	5112
3.20	5681	1.428	.207	1.635	1.517	288.7	267.9	6463	5130
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.41	5900	1.431	.192	1.624	1.506	297.7	276.2	6352	4585

TABLE III-17 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 500 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.483	.145	1.628	1.481	291.0	264.7	5460	2810
2.00	5872	1.486	.153	1.639	1.493	299.2	272.4	5911	3333
2.20	5914	1.488	.165	1.652	1.506	303.8	276.8	6187	3870
2.40	5901	1.488	.177	1.665	1.518	305.5	278.5	6346	4370
2.60	5856	1.487	.187	1.674	1.527	304.7	277.9	6426	4738
2.80	5804	1.486	.190	1.676	1.529	302.4	275.9	6464	4926
3.00	5743	1.486	.192	1.678	1.531	299.5	273.3	6472	5000
3.20	5681	1.486	.193	1.679	1.532	296.5	270.5	6463	5023
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.44	5895	1.488	.179	1.667	1.520	305.5	278.6	6367	4447

PC= 500 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.520	.134	1.654	1.478	295.6	264.0	5460	2666
2.00	5872	1.524	.143	1.667	1.490	304.2	272.0	5911	3173
2.20	5914	1.529	.151	1.679	1.503	308.7	276.3	6187	3700
2.40	5901	1.531	.163	1.694	1.518	310.7	278.4	6346	4210
2.60	5856	1.531	.173	1.704	1.528	310.2	278.1	6426	4612
2.80	5804	1.530	.180	1.710	1.534	308.5	276.7	6464	4827
3.00	5743	1.530	.182	1.712	1.535	305.6	274.1	6472	4912
3.20	5681	1.530	.183	1.713	1.537	302.5	271.3	6463	4938
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	5891	1.532	.166	1.698	1.521	310.9	278.6	6379	4344

PC= 500 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.548	.126	1.674	1.468	299.1	262.4	5460	2552
2.00	5872	1.553	.134	1.688	1.482	308.0	270.5	5911	3045
2.20	5914	1.560	.145	1.705	1.499	313.4	275.6	6187	3564
2.40	5901	1.564	.158	1.722	1.516	315.8	278.1	6346	4075
2.60	5856	1.565	.169	1.734	1.528	315.6	278.1	6426	4504
2.80	5804	1.565	.172	1.737	1.532	313.4	276.3	6464	4745
3.00	5743	1.565	.175	1.740	1.534	310.5	273.8	6472	4839
3.20	5681	1.565	.176	1.741	1.535	307.4	271.1	6463	4869
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.48	5887	1.565	.162	1.727	1.521	316.0	278.4	6388	4259

PC= 500 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.570	.121	1.691	1.456	302.2	260.2	5460	2460
2.00	5872	1.577	.129	1.706	1.471	311.4	268.5	5911	2939
2.20	5914	1.586	.136	1.722	1.487	316.6	273.4	6187	3447
2.40	5901	1.592	.148	1.741	1.505	319.3	276.1	6346	3959
2.60	5856	1.594	.160	1.754	1.519	319.3	276.5	6426	4408
2.80	5804	1.594	.166	1.760	1.525	317.5	275.1	6464	4673
3.00	5743	1.594	.169	1.763	1.528	314.7	272.7	6472	4778
3.20	5681	1.594	.170	1.764	1.529	311.5	270.0	6463	4811
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.50	5879	1.593	.155	1.748	1.513	319.5	276.5	6392	4208

TABLE III-17 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 500 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 10000.0000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.606	.111	1.717	1.423	306.8	254.2	5400	2313
2.00	5872	1.615	.117	1.732	1.438	316.2	262.5	5911	2768
2.20	5914	1.626	.127	1.753	1.459	322.2	268.2	6187	3262
2.40	5901	1.635	.138	1.773	1.479	325.3	271.4	6346	3769
2.60	5856	1.639	.150	1.789	1.495	325.7	272.2	6426	4243
2.80	5804	1.639	.158	1.797	1.503	324.1	271.1	6464	4552
3.00	5743	1.639	.161	1.800	1.506	321.3	268.8	6472	4678
3.20	5681	1.640	.162	1.802	1.508	318.1	266.2	6463	4718
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.54	5870	1.638	.147	1.785	1.491	325.8	272.1	6406	4114

PC= 500 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.633	.103	1.735	1.383	310.1	247.1	5400	2200
2.00	5872	1.644	.110	1.753	1.400	320.0	255.6	5911	2639
2.20	5914	1.656	.119	1.775	1.423	326.4	261.5	6187	3118
2.40	5901	1.668	.130	1.798	1.445	329.8	265.1	6346	3617
2.60	5856	1.674	.141	1.814	1.462	330.3	266.1	6426	4105
2.80	5804	1.675	.149	1.823	1.471	329.0	265.3	6464	4450
3.00	5743	1.675	.152	1.827	1.475	326.2	263.2	6472	4596
3.20	5681	1.675	.154	1.829	1.477	323.0	260.7	6463	4642
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.55	5867	1.673	.138	1.811	1.459	330.3	266.0	6411	4005

PC= 500 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.662	.095	1.758	1.317	314.1	235.4	5400	2074
2.00	5872	1.675	.102	1.777	1.336	324.4	243.9	5911	2491
2.20	5914	1.690	.111	1.801	1.360	331.1	250.1	6187	2951
2.40	5901	1.705	.122	1.826	1.385	335.0	254.1	6346	3438
2.60	5856	1.714	.131	1.845	1.404	335.8	255.6	6426	3933
2.80	5804	1.715	.141	1.856	1.415	334.8	255.3	6464	4323
3.00	5743	1.716	.145	1.861	1.420	332.2	253.4	6472	4498
3.20	5681	1.716	.146	1.863	1.422	329.0	251.1	6463	4552
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.59	5858	1.713	.131	1.844	1.403	335.8	255.5	6424	3914

PC= 500 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.697	.087	1.784	1.197	318.9	213.8	5400	1925
2.00	5872	1.712	.093	1.805	1.217	329.5	222.2	5911	2315
2.20	5914	1.731	.101	1.831	1.243	336.7	228.6	6187	2749
2.40	5901	1.749	.110	1.859	1.271	341.0	233.2	6346	3217
2.60	5856	1.761	.121	1.882	1.294	342.5	235.5	6426	3710
2.80	5804	1.764	.131	1.895	1.308	342.0	235.9	6464	4147
3.00	5743	1.765	.136	1.901	1.314	339.4	234.5	6472	4372
3.20	5681	1.766	.138	1.904	1.316	336.2	232.4	6463	4440
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.65	5844	1.762	.123	1.886	1.298	342.5	235.8	6439	3818

TABLE III-17 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 500 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.741	.077	1.818	.936	324.8	167.2	5460	1739
2.00	5872	1.759	.082	1.841	.959	335.9	175.0	5911	2095
2.20	5914	1.781	.088	1.869	.988	343.7	181.6	6187	2491
2.40	5901	1.804	.097	1.901	1.019	348.7	186.9	6346	2928
2.60	5856	1.821	.108	1.928	1.047	351.0	190.5	6426	3405
2.80	5804	1.828	.118	1.946	1.064	351.1	192.0	6464	3880
3.00	5743	1.829	.125	1.954	1.073	348.9	191.5	6472	4190
3.20	5681	1.830	.127	1.958	1.076	345.7	190.0	6463	4287
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.70	5830	1.825	.113	1.939	1.057	351.3	191.5	6450	3674

PC= 500 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.769	.072	1.840	.665	328.9	118.8	5460	1623
2.00	5872	1.788	.075	1.863	.688	340.1	125.5	5911	1955
2.20	5914	1.813	.081	1.894	.718	348.2	132.1	6187	2326
2.40	5901	1.838	.089	1.928	.752	353.6	137.9	6346	2739
2.60	5856	1.859	.099	1.958	.782	356.4	142.4	6426	3199
2.80	5804	1.869	.109	1.979	.803	357.0	144.9	6464	3680
3.00	5743	1.872	.117	1.989	.813	355.1	145.2	6472	4053
3.20	5681	1.873	.120	1.993	.817	351.9	144.3	6463	4180
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.75	5818	1.867	.107	1.974	.799	357.1	144.4	6457	3563

PC= 500 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5749	1.787	.080	1.867	.397	333.6	71.0	5460	1549
2.00	5872	1.807	.080	1.887	.417	344.4	76.2	5911	1860
2.20	5914	1.835	.077	1.912	.442	351.5	81.3	6187	2208
2.40	5901	1.863	.083	1.946	.477	357.0	87.4	6346	2601
2.60	5856	1.887	.092	1.979	.509	360.2	92.7	6426	3046
2.80	5804	1.899	.103	2.002	.532	361.2	96.1	6464	3522
3.00	5743	1.903	.112	2.015	.545	359.6	97.3	6472	3939
3.20	5681	1.904	.115	2.019	.550	356.6	97.1	6463	4098
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	5810	1.898	.102	2.000	.530	361.2	95.8	6461	3469

TABLE III-18

LOX/RP-1, P_c = 600

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 600 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	.682	.561	1.244	1.219	222.3	218.0	5476	4979
2.00	5878	.671	.568	1.239	1.214	226.4	221.9	5937	5497
2.20	5926	.663	.573	1.236	1.211	227.6	223.1	6231	5852
2.40	5915	.658	.576	1.234	1.209	226.9	222.4	6395	6058
2.60	5873	.655	.578	1.233	1.208	225.0	220.6	6478	6163
2.80	5817	.653	.579	1.232	1.208	222.8	218.3	6513	6211
3.00	5756	.653	.579	1.232	1.207	220.4	216.0	6522	6226
3.20	5695	.652	.580	1.232	1.207	218.0	213.7	6513	6222
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.22	5927	.662	.574	1.235	1.211	227.6	223.1	6257	5884

PC= 600 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.230	.234	1.464	1.415	261.8	253.0	5476	3708
2.00	5878	1.222	.245	1.467	1.418	268.0	259.1	5937	4292
2.20	5926	1.214	.256	1.471	1.422	270.9	261.9	6231	4807
2.40	5915	1.208	.266	1.474	1.425	271.0	262.0	6395	5185
2.60	5873	1.204	.271	1.476	1.427	269.4	260.5	6478	5397
2.80	5817	1.203	.274	1.477	1.428	267.0	258.1	6513	5494
3.00	5756	1.201	.275	1.477	1.428	264.2	255.4	6522	5534
3.20	5695	1.201	.276	1.477	1.428	261.4	252.7	6513	5542
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.32	5923	1.210	.262	1.473	1.424	271.2	262.2	6335	5045

PC= 600 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.361	.186	1.547	1.473	276.6	263.4	5476	3269
2.00	5878	1.357	.196	1.553	1.480	283.8	270.4	5937	3833
2.20	5926	1.354	.208	1.561	1.488	287.6	274.1	6231	4372
2.40	5915	1.352	.205	1.557	1.484	286.3	272.8	6395	4825
2.60	5873	1.349	.212	1.561	1.488	285.0	271.6	6478	5105
2.80	5817	1.347	.215	1.563	1.489	282.6	269.3	6513	5237
3.00	5756	1.347	.217	1.563	1.490	279.7	266.6	6522	5291
3.20	5695	1.346	.217	1.564	1.490	276.8	263.8	6513	5306
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.25	5928	1.353	.208	1.561	1.488	287.7	274.2	6284	4494

PC= 600 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.435	.160	1.595	1.497	285.1	267.6	5476	2999
2.00	5878	1.434	.169	1.603	1.505	292.9	275.0	5937	3540
2.20	5926	1.434	.179	1.613	1.515	297.2	279.1	6231	4080
2.40	5915	1.433	.191	1.623	1.525	298.5	280.5	6395	4569
2.60	5873	1.430	.200	1.630	1.532	297.5	279.6	6478	4906
2.80	5817	1.429	.204	1.632	1.534	295.2	277.5	6513	5071
3.00	5756	1.428	.206	1.634	1.536	292.3	274.8	6522	5137
3.20	5695	1.428	.206	1.634	1.536	289.3	272.0	6513	5157
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.42	5913	1.432	.192	1.624	1.526	298.5	280.5	6404	4600

TABLE III-18 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 600 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.484	.145	1.628	1.506	291.1	269.2	5476	2809
2.00	5878	1.486	.153	1.639	1.516	299.4	277.0	5937	3330
2.20	5926	1.488	.164	1.652	1.530	304.3	281.7	6231	3862
2.40	5915	1.489	.176	1.665	1.542	306.1	283.6	6395	4369
2.60	5873	1.488	.185	1.672	1.550	305.3	282.9	6478	4751
2.80	5817	1.486	.190	1.676	1.554	303.0	280.9	6513	4945
3.00	5756	1.486	.191	1.678	1.555	300.2	278.2	6522	5024
3.20	5695	1.486	.192	1.678	1.556	297.1	275.4	6513	5047
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.44	5909	1.489	.178	1.667	1.544	306.1	283.6	6418	4454

PC= 600 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.520	.134	1.654	1.507	295.7	269.4	5476	2666
2.00	5878	1.524	.142	1.666	1.519	304.4	277.5	5937	3170
2.20	5926	1.528	.153	1.681	1.534	309.6	282.5	6231	3691
2.40	5915	1.531	.164	1.696	1.549	311.8	284.8	6395	4205
2.60	5873	1.531	.174	1.705	1.558	311.3	284.4	6478	4620
2.80	5817	1.530	.179	1.710	1.563	309.1	282.5	6513	4844
3.00	5756	1.530	.181	1.711	1.565	306.2	279.9	6522	4935
3.20	5695	1.530	.182	1.712	1.565	303.1	277.1	6513	4962
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	5905	1.532	.167	1.699	1.552	311.9	285.0	6428	4339

PC= 600 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.548	.126	1.674	1.502	299.2	268.6	5476	2552
2.00	5878	1.553	.134	1.687	1.516	308.2	276.9	5937	3041
2.20	5926	1.560	.144	1.704	1.532	313.8	282.3	6231	3552
2.40	5915	1.565	.156	1.720	1.549	316.3	284.8	6395	4068
2.60	5873	1.566	.166	1.731	1.560	316.1	284.8	6478	4508
2.80	5817	1.565	.172	1.737	1.565	314.0	283.0	6513	4759
3.00	5756	1.565	.174	1.739	1.568	311.2	280.5	6522	4861
3.20	5695	1.565	.175	1.740	1.569	308.0	277.7	6513	4893
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.48	5901	1.566	.160	1.726	1.554	316.6	285.1	6438	4255

PC= 600 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.570	.120	1.691	1.495	302.3	267.3	5476	2459
2.00	5878	1.577	.128	1.705	1.509	311.6	275.7	5937	2935
2.20	5926	1.586	.136	1.722	1.526	317.2	281.1	6231	3435
2.40	5915	1.592	.148	1.740	1.544	319.9	283.9	6395	3950
2.60	5873	1.594	.159	1.753	1.557	320.0	284.3	6478	4410
2.80	5817	1.594	.166	1.760	1.564	318.2	282.7	6513	4686
3.00	5756	1.594	.169	1.762	1.566	315.3	280.2	6522	4798
3.20	5695	1.594	.170	1.763	1.567	312.1	277.5	6513	4833
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.51	5893	1.594	.155	1.748	1.552	320.2	284.3	6447	4225

TABLE III-18 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60 .

LO2/RP-1 12-27-57

PC= 600 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.606	.111	1.716	1.472	306.9	263.1	5476	2311
2.00	5878	1.615	.117	1.731	1.487	316.4	271.6	5937	2766
2.20	5926	1.626	.126	1.751	1.507	322.6	277.5	6231	3248
2.40	5915	1.635	.137	1.772	1.527	325.9	280.8	6395	3757
2.60	5873	1.639	.149	1.788	1.543	326.4	281.7	6478	4239
2.80	5817	1.639	.157	1.796	1.551	324.7	280.4	6513	4562
3.00	5756	1.639	.160	1.799	1.555	322.0	278.1	6522	4695
3.20	5695	1.639	.162	1.801	1.556	318.8	275.4	6513	4738
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.55	5885	1.638	.146	1.785	1.540	326.5	281.7	6461	4130

PC= 600 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.633	.102	1.735	1.441	310.2	257.7	5476	2198
2.00	5878	1.643	.109	1.752	1.458	320.2	266.5	5937	2636
2.20	5926	1.656	.118	1.774	1.480	326.8	272.6	6231	3104
2.40	5915	1.667	.129	1.797	1.503	330.3	276.3	6395	3603
2.60	5873	1.674	.140	1.813	1.519	331.0	277.4	6478	4097
2.80	5817	1.674	.148	1.823	1.529	329.6	276.4	6513	4458
3.00	5756	1.675	.152	1.827	1.533	326.9	274.3	6522	4612
3.20	5695	1.675	.153	1.828	1.534	323.6	271.6	6513	4662
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.56	5881	1.673	.138	1.811	1.517	331.1	277.3	6466	4016

PC= 600 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.662	.095	1.758	1.390	314.3	248.6	5476	2072
2.00	5878	1.674	.102	1.776	1.409	324.5	257.4	5937	2488
2.20	5926	1.690	.110	1.800	1.432	331.5	263.8	6231	2938
2.40	5915	1.704	.120	1.825	1.457	335.5	267.9	6395	3422
2.60	5873	1.713	.130	1.843	1.476	336.5	269.4	6478	3921
2.80	5817	1.715	.140	1.855	1.488	335.4	269.0	6513	4325
3.00	5756	1.716	.144	1.860	1.493	332.8	267.1	6522	4512
3.20	5695	1.716	.146	1.862	1.494	329.6	264.5	6513	4569
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.60	5874	1.713	.130	1.843	1.476	336.5	269.4	6477	3912

PC= 600 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.698	.087	1.784	1.295	319.1	231.5	5476	1922
2.00	5878	1.712	.092	1.804	1.314	329.6	240.1	5937	2312
2.20	5926	1.730	.100	1.829	1.340	337.0	246.8	6231	2736
2.40	5915	1.748	.109	1.857	1.367	341.5	251.4	6395	3200
2.60	5873	1.760	.119	1.880	1.390	343.1	253.7	6478	3694
2.80	5817	1.764	.130	1.894	1.404	342.5	253.9	6513	4144
3.00	5756	1.765	.136	1.901	1.411	340.0	252.4	6522	4382
3.20	5695	1.765	.138	1.903	1.413	336.9	250.2	6513	4456
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.64	5862	1.762	.122	1.883	1.394	343.2	253.9	6490	3797

TABLE III-18 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 600 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.741	.077	1.818	1.083	325.1	193.7	5476	1734
2.00	5878	1.758	.082	1.839	1.105	336.1	201.8	5937	2091
2.20	5926	1.780	.088	1.867	1.132	344.0	208.6	6231	2479
2.40	5915	1.802	.096	1.899	1.164	349.1	214.0	6395	2911
2.60	5873	1.820	.106	1.926	1.191	351.6	217.5	6478	3387
2.80	5817	1.827	.117	1.944	1.210	351.6	218.7	6513	3869
3.00	5756	1.829	.125	1.953	1.219	349.5	218.0	6522	4194
3.20	5695	1.829	.127	1.956	1.222	346.3	216.3	6513	4300
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.70	5847	1.825	.112	1.936	1.201	351.9	218.3	6501	3620

PC= 600 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.769	.072	1.841	.861	329.2	154.0	5476	1617
2.00	5878	1.787	.075	1.862	.882	340.2	161.2	5937	1952
2.20	5926	1.811	.081	1.892	.912	348.4	168.0	6231	2314
2.40	5915	1.837	.088	1.925	.945	354.0	173.8	6395	2723
2.60	5873	1.857	.098	1.955	.975	356.9	178.1	6478	3179
2.80	5817	1.868	.109	1.977	.997	357.5	180.3	6513	3666
3.00	5756	1.871	.117	1.988	1.008	355.6	180.3	6522	4053
3.20	5695	1.872	.120	1.992	1.012	352.6	179.1	6513	4191
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.74	5833	1.866	.106	1.972	.992	357.5	179.9	6506	3542

PC= 600 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5752	1.788	.080	1.867	.643	333.9	114.9	5476	1542
2.00	5878	1.806	.080	1.886	.661	344.6	120.8	5937	1862
2.20	5926	1.832	.082	1.914	.689	352.6	127.0	6231	2204
2.40	5915	1.861	.086	1.947	.722	357.9	132.8	6395	2589
2.60	5873	1.885	.093	1.977	.752	360.9	137.4	6478	3028
2.80	5817	1.898	.102	2.000	.775	361.7	140.2	6513	3508
3.00	5756	1.902	.111	2.013	.789	360.2	141.1	6522	3935
3.20	5695	1.903	.115	2.018	.793	357.2	140.4	6513	4107
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.77	5826	1.897	.100	1.997	.772	361.7	139.9	6509	3430

TABLE III-19

LOX/RP-1, P_c = 700

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 700 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	.683	.561	1.244	1.223	222.4	218.7	5487	4984
2.00	5882	.672	.567	1.239	1.218	226.6	222.8	5958	5510
2.20	5927	.663	.573	1.236	1.215	227.6	223.8	6252	5865
2.40	5920	.656	.577	1.234	1.213	227.0	223.1	6425	6082
2.60	5881	.655	.577	1.232	1.211	225.2	221.4	6513	6193
2.80	5828	.653	.579	1.232	1.211	223.2	219.4	6556	6248
3.00	5767	.653	.579	1.232	1.211	220.8	217.1	6564	6264
3.20	5706	.652	.579	1.232	1.211	218.4	214.7	6556	6260
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.22	5929	.662	.573	1.235	1.214	227.7	223.8	6278	5897

PC= 700 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.231	.234	1.464	1.422	261.9	254.4	5487	3707
2.00	5882	1.223	.244	1.467	1.425	268.2	260.5	5958	4291
2.20	5927	1.214	.256	1.470	1.428	270.9	263.1	6252	4804
2.40	5920	1.208	.265	1.473	1.431	271.1	263.4	6425	5190
2.60	5881	1.204	.271	1.475	1.433	269.6	261.9	6513	5412
2.80	5828	1.203	.274	1.476	1.434	267.5	259.9	6556	5520
3.00	5767	1.202	.275	1.477	1.435	264.7	257.2	6564	5561
3.20	5706	1.201	.275	1.477	1.435	261.9	254.5	6556	5571
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.33	5926	1.210	.263	1.472	1.431	271.3	263.5	6374	5075

PC= 700 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.361	.186	1.547	1.484	276.6	265.3	5487	3269
2.00	5882	1.357	.196	1.553	1.490	283.9	272.4	5958	3831
2.20	5927	1.353	.208	1.561	1.498	287.6	276.0	6252	4369
2.40	5920	1.352	.205	1.556	1.493	286.4	274.8	6425	4824
2.60	5881	1.348	.212	1.560	1.497	285.2	273.7	6513	5115
2.80	5828	1.348	.215	1.563	1.500	283.1	271.7	6556	5259
3.00	5767	1.347	.216	1.563	1.500	280.3	269.0	6564	5315
3.20	5706	1.346	.217	1.564	1.501	277.3	266.2	6556	5331
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.25	5930	1.352	.209	1.561	1.498	287.8	276.2	6307	4493

PC= 700 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.434	.160	1.594	1.510	285.1	270.1	5487	2998
2.00	5882	1.434	.169	1.603	1.519	293.0	277.7	5958	3537
2.20	5927	1.434	.178	1.612	1.528	296.9	281.5	6252	4076
2.40	5920	1.432	.190	1.622	1.538	298.5	283.0	6425	4565
2.60	5881	1.430	.199	1.629	1.545	297.7	282.4	6513	4912
2.80	5828	1.429	.203	1.632	1.548	295.7	280.5	6556	5089
3.00	5767	1.428	.205	1.634	1.550	292.8	277.8	6564	5160
3.20	5706	1.428	.206	1.634	1.550	289.8	274.9	6556	5180
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.43	5916	1.432	.192	1.623	1.539	298.5	283.1	6445	4631

TABLE III-19 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE CQ3 29SEPT, 60

LO2/RP-1 12-27-57

PC= 700 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.483	.145	1.628	1.523	291.2	272.4	5487	2808
2.00	5882	1.495	.153	1.638	1.533	299.5	280.3	5958	3327
2.20	5927	1.488	.164	1.651	1.546	304.2	284.9	6252	3859
2.40	5920	1.488	.176	1.664	1.559	306.2	286.9	6425	4363
2.60	5881	1.487	.185	1.672	1.567	305.7	286.5	6513	4753
2.80	5828	1.487	.189	1.676	1.571	303.6	284.6	6556	4962
3.00	5767	1.486	.191	1.677	1.572	300.7	281.9	6564	5045
3.20	5706	1.486	.192	1.678	1.573	297.6	279.0	6556	5070
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.46	5912	1.488	.179	1.667	1.562	306.3	287.0	6460	4489

PC= 700 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.519	.134	1.653	1.527	295.7	273.2	5487	2664
2.00	5882	1.523	.142	1.665	1.539	304.4	281.4	5958	3167
2.20	5927	1.527	.154	1.681	1.556	309.8	286.6	6252	3691
2.40	5920	1.531	.161	1.692	1.566	311.4	288.3	6425	4199
2.60	5881	1.531	.172	1.702	1.576	311.2	288.2	6513	4620
2.80	5828	1.530	.179	1.709	1.583	309.6	286.8	6556	4858
3.00	5767	1.530	.181	1.711	1.585	306.7	284.2	6564	4954
3.20	5706	1.530	.182	1.712	1.586	303.6	281.3	6556	4983
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.40	5909	1.532	.165	1.696	1.570	311.6	288.4	6468	4369

PC= 700 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.548	.126	1.673	1.526	299.3	273.0	5487	2551
2.00	5882	1.553	.133	1.686	1.539	308.3	281.5	5958	3037
2.20	5927	1.559	.144	1.703	1.556	313.8	286.7	6252	3551
2.40	5920	1.564	.156	1.720	1.573	316.5	289.4	6425	4062
2.60	5881	1.564	.167	1.731	1.584	316.5	289.6	6513	4506
2.80	5828	1.565	.171	1.736	1.589	314.5	287.9	6556	4772
3.00	5767	1.565	.174	1.739	1.592	311.7	285.4	6564	4879
3.20	5706	1.565	.175	1.740	1.593	308.6	282.5	6556	4912
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.50	5901	1.564	.163	1.727	1.580	316.7	289.8	6476	4314

PC= 700 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.570	.120	1.691	1.523	302.4	272.3	5487	2457
2.00	5882	1.577	.128	1.704	1.536	311.6	280.9	5958	2932
2.20	5927	1.585	.135	1.720	1.552	317.0	286.0	6252	3434
2.40	5920	1.592	.147	1.738	1.570	319.9	289.0	6425	3944
2.60	5881	1.593	.158	1.752	1.584	320.2	289.5	6513	4405
2.80	5828	1.594	.165	1.759	1.591	318.7	288.2	6556	4696
3.00	5767	1.593	.168	1.762	1.594	315.8	285.7	6564	4815
3.20	5706	1.594	.169	1.763	1.595	312.7	282.9	6556	4853
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.53	5895	1.593	.155	1.748	1.580	320.3	289.5	6488	4267

TABLE III-19 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 700 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.606	.111	1.716	1.506	306.9	269.4	5487	2310
2.00	5882	1.614	.116	1.731	1.521	316.4	278.0	5958	2763
2.20	5927	1.625	.126	1.751	1.541	322.6	283.9	6252	3250
2.40	5920	1.634	.137	1.771	1.561	325.9	287.3	6425	3752
2.60	5881	1.638	.148	1.786	1.576	326.5	288.2	6513	4233
2.80	5828	1.639	.157	1.795	1.585	325.2	287.2	6556	4570
3.00	5767	1.639	.160	1.799	1.589	322.5	284.8	6564	4711
3.20	5706	1.639	.161	1.800	1.590	319.3	282.1	6556	4755
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.57	5888	1.637	.147	1.784	1.574	326.6	288.1	6501	4160

PC= 700 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.632	.102	1.735	1.483	310.3	265.2	5487	2197
2.00	5882	1.642	.109	1.751	1.500	320.2	274.2	5958	2633
2.20	5927	1.655	.118	1.773	1.521	326.7	280.3	6252	3106
2.40	5920	1.666	.129	1.795	1.543	330.4	284.0	6425	3600
2.60	5881	1.672	.139	1.811	1.560	331.1	285.1	6513	4090
2.80	5828	1.674	.148	1.822	1.570	330.0	284.4	6556	4462
3.00	5767	1.674	.152	1.826	1.574	327.4	282.2	6564	4626
3.20	5706	1.675	.153	1.828	1.576	324.2	279.5	6556	4678
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.58	5884	1.672	.138	1.810	1.558	331.1	285.1	6507	4052

PC= 700 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.662	.095	1.757	1.442	314.3	258.0	5487	2071
2.00	5882	1.674	.102	1.775	1.461	324.6	267.0	5958	2486
2.20	5927	1.689	.110	1.799	1.484	331.5	273.4	6252	2940
2.40	5920	1.703	.120	1.823	1.508	335.5	277.6	6425	3420
2.60	5881	1.712	.130	1.841	1.527	336.6	279.1	6513	3913
2.80	5828	1.715	.139	1.854	1.539	335.9	278.8	6556	4325
3.00	5767	1.715	.144	1.859	1.544	333.3	276.9	6564	4524
3.20	5706	1.715	.146	1.861	1.546	330.1	274.3	6556	4586
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.62	5876	1.713	.131	1.843	1.528	336.6	279.1	6520	3959

PC= 700 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.697	.087	1.784	1.364	319.1	244.0	5487	1921
2.00	5882	1.711	.092	1.803	1.383	329.7	252.9	5958	2310
2.20	5927	1.729	.100	1.829	1.409	336.9	259.6	6252	2739
2.40	5920	1.746	.109	1.856	1.436	341.5	264.2	6425	3199
2.60	5881	1.759	.119	1.878	1.458	343.2	266.5	6513	3688
2.80	5828	1.764	.129	1.893	1.473	342.9	266.9	6556	4140
3.00	5767	1.764	.136	1.900	1.480	340.6	265.3	6564	4391
3.20	5706	1.765	.137	1.902	1.483	337.4	262.9	6556	4470
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.67	5863	1.761	.123	1.884	1.464	343.4	266.8	6534	3853

TABLE III-19 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 700 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.741	.077	1.818	1.188	325.1	212.5	5487	1733
2.00	5882	1.757	.081	1.839	1.209	336.2	221.0	5958	2089
2.20	5927	1.779	.088	1.867	1.237	343.9	227.9	6252	2482
2.40	5920	1.801	.096	1.897	1.267	349.1	233.2	6425	2911
2.60	5881	1.818	.106	1.924	1.294	351.7	236.5	6513	3382
2.80	5828	1.827	.116	1.943	1.313	352.0	237.9	6556	3860
3.00	5767	1.828	.124	1.952	1.323	350.0	237.1	6564	4199
3.20	5706	1.829	.127	1.956	1.326	346.9	235.2	6556	4311
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.73	5848	1.824	.113	1.937	1.307	352.1	237.6	6544	3700

PC= 700 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.769	.072	1.841	1.001	329.2	179.0	5487	1613
2.00	5882	1.786	.075	1.861	1.021	340.3	186.8	5958	1949
2.20	5927	1.810	.081	1.891	1.051	348.4	193.7	6252	2318
2.40	5920	1.835	.089	1.924	1.084	354.0	199.5	6425	2724
2.60	5881	1.855	.097	1.953	1.113	357.0	203.5	6513	3176
2.80	5828	1.868	.108	1.975	1.136	357.9	205.7	6556	3653
3.00	5767	1.870	.116	1.987	1.147	356.1	205.6	6564	4052
3.20	5706	1.871	.119	1.991	1.151	353.1	204.2	6556	4200
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.77	5837	1.866	.106	1.972	1.133	357.9	205.5	6551	3579

PC= 700 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.787	.080	1.867	.817	333.9	146.1	5487	1533
2.00	5882	1.805	.080	1.885	.835	344.7	152.7	5958	1856
2.20	5927	1.833	.076	1.909	.859	351.7	158.3	6252	2200
2.40	5920	1.860	.082	1.942	.892	357.4	164.2	6425	2587
2.60	5881	1.883	.091	1.974	.924	360.8	168.9	6513	3024
2.80	5828	1.897	.101	1.998	.949	362.0	171.9	6556	3495
3.00	5767	1.901	.111	2.012	.962	360.7	172.5	6564	3931
3.20	5706	1.902	.115	2.017	.967	357.7	171.6	6556	4114
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.80	5829	1.897	.101	1.998	.948	362.0	171.8	6555	3484

TABLE III-20

LOX/RP-1, P_c = 800

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 800 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	.683	.561	1.244	1.225	222.5	219.2	5496	4988
2.00	5885	.672	.567	1.239	1.221	226.7	223.4	5976	5521
2.20	5939	.663	.573	1.236	1.218	228.2	224.8	6287	5894
2.40	5933	.658	.576	1.234	1.216	227.6	224.2	6463	6114
2.60	5893	.655	.578	1.233	1.215	225.8	222.5	6553	6228
2.80	5838	.654	.579	1.232	1.214	223.6	220.3	6592	6281
3.00	5777	.653	.579	1.232	1.213	221.2	217.9	6602	6297
3.20	5716	.652	.579	1.232	1.213	218.8	215.6	6593	6294
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.24	5942	.662	.573	1.236	1.217	228.2	224.8	6333	5950

PC= 800 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.231	.233	1.464	1.427	261.9	255.3	5496	3706
2.00	5885	1.223	.244	1.467	1.430	268.3	261.6	5976	4290
2.20	5939	1.215	.255	1.470	1.434	271.4	264.6	6287	4812
2.40	5933	1.209	.265	1.474	1.437	271.8	265.0	6463	5207
2.60	5893	1.205	.271	1.475	1.439	270.3	263.5	6553	5436
2.80	5838	1.203	.273	1.476	1.440	267.9	261.2	6592	5542
3.00	5777	1.202	.275	1.477	1.440	265.2	258.6	6602	5586
3.20	5716	1.202	.275	1.477	1.440	262.4	255.8	6593	5596
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.34	5938	1.211	.262	1.473	1.436	271.9	265.1	6418	5102

PC= 800 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.361	.186	1.546	1.491	276.7	266.8	5496	3268
2.00	5885	1.357	.195	1.553	1.498	284.0	274.0	5976	3827
2.20	5939	1.354	.206	1.560	1.505	288.1	277.9	6287	4367
2.40	5933	1.352	.204	1.556	1.501	287.0	276.9	6463	4833
2.60	5893	1.349	.211	1.560	1.505	285.8	275.7	6553	5133
2.80	5838	1.348	.215	1.562	1.507	283.5	273.5	6592	5277
3.00	5777	1.347	.216	1.563	1.508	280.7	270.8	6602	5337
3.20	5716	1.347	.217	1.563	1.508	277.8	268.0	6593	5354
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.26	5943	1.353	.207	1.560	1.505	288.3	278.1	6353	4513

PC= 800 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.434	.160	1.594	1.521	285.2	272.0	5496	2996
2.00	5885	1.434	.168	1.602	1.529	293.1	279.7	5976	3535
2.20	5939	1.434	.178	1.612	1.538	297.6	284.0	6287	4071
2.40	5933	1.433	.190	1.622	1.549	299.1	285.6	6463	4568
2.60	5893	1.430	.198	1.629	1.555	298.4	284.9	6553	4925
2.80	5838	1.429	.203	1.632	1.558	296.1	282.8	6592	5105
3.00	5777	1.428	.205	1.633	1.560	293.3	280.1	6602	5179
3.20	5716	1.428	.206	1.634	1.560	290.3	277.2	6593	5201
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.43	5928	1.432	.191	1.623	1.550	299.2	285.6	6484	4637

TABLE III-20 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 800 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.483	.144	1.628	1.536	291.2	274.8	5496	2808
2.00	5885	1.485	.153	1.638	1.546	299.6	282.8	5976	3325
2.20	5939	1.488	.163	1.650	1.559	304.7	287.7	6287	3852
2.40	5933	1.489	.174	1.663	1.571	306.7	289.8	6463	4363
2.60	5893	1.488	.184	1.671	1.579	306.2	289.3	6553	4763
2.80	5838	1.486	.189	1.675	1.583	304.0	287.3	6592	4975
3.00	5777	1.486	.191	1.677	1.585	301.2	284.7	6602	5062
3.20	5716	1.486	.192	1.678	1.586	298.1	281.8	6593	5088
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.46	5925	1.489	.177	1.666	1.574	306.8	289.9	6497	4487

PC= 800 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.519	.134	1.653	1.543	295.8	276.0	5496	2663
2.00	5885	1.523	.142	1.665	1.555	304.5	284.4	5976	3164
2.20	5939	1.528	.151	1.679	1.569	310.0	289.6	6287	3680
2.40	5933	1.531	.163	1.694	1.584	312.4	292.1	6463	4196
2.60	5893	1.531	.173	1.704	1.593	312.1	291.9	6553	4626
2.80	5838	1.530	.179	1.708	1.598	310.0	290.0	6592	4871
3.00	5777	1.530	.181	1.711	1.601	307.2	287.4	6602	4970
3.20	5716	1.530	.182	1.712	1.601	304.1	284.5	6593	5002
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.48	5922	1.531	.167	1.698	1.588	312.6	292.3	6507	4369

PC= 800 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.548	.126	1.673	1.545	299.3	276.3	5496	2549
2.00	5885	1.553	.133	1.686	1.557	308.4	284.9	5976	3036
2.20	5939	1.559	.143	1.702	1.573	314.2	290.4	6287	3542
2.40	5933	1.564	.154	1.718	1.590	316.9	293.2	6463	4055
2.60	5893	1.566	.164	1.730	1.601	316.9	293.3	6553	4509
2.80	5838	1.565	.171	1.736	1.607	315.0	291.7	6592	4782
3.00	5777	1.565	.174	1.738	1.610	312.2	289.1	6602	4894
3.20	5716	1.564	.175	1.739	1.611	309.0	286.2	6593	4930
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.50	5917	1.565	.159	1.725	1.596	317.2	293.6	6517	4283

PC= 800 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.570	.120	1.690	1.543	302.4	276.1	5496	2456
2.00	5885	1.576	.128	1.704	1.557	311.7	284.8	5976	2930
2.20	5939	1.585	.135	1.720	1.573	317.5	290.3	6287	3425
2.40	5933	1.592	.146	1.738	1.591	320.5	293.4	6463	3936
2.60	5893	1.594	.158	1.751	1.604	320.8	293.9	6553	4406
2.80	5838	1.593	.165	1.758	1.611	319.1	292.4	6592	4706
3.00	5777	1.593	.168	1.761	1.614	316.3	289.9	6602	4829
3.20	5716	1.593	.169	1.762	1.615	313.1	287.0	6593	4869
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.53	5908	1.593	.154	1.747	1.600	320.9	294.0	6527	4260

TABLE III-20 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 800 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.606	.110	1.716	1.532	307.0	274.1	5496	2309
2.00	5885	1.614	.116	1.730	1.546	316.5	282.9	5976	2760
2.20	5939	1.624	.125	1.749	1.566	322.9	289.0	6287	3238
2.40	5933	1.634	.136	1.770	1.586	326.4	292.5	6463	3742
2.60	5893	1.638	.148	1.786	1.602	327.2	293.5	6553	4229
2.80	5838	1.639	.156	1.795	1.611	325.7	292.3	6592	4575
3.00	5777	1.639	.160	1.798	1.615	322.9	289.9	6602	4723
3.20	5716	1.639	.161	1.800	1.616	319.8	287.1	6593	4770
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.57	5900	1.638	.146	1.784	1.600	327.2	293.5	6542	4162

PC= 800 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.632	.102	1.735	1.514	310.3	270.9	5496	2197
2.00	5885	1.642	.109	1.751	1.531	320.3	280.0	5976	2631
2.20	5939	1.654	.117	1.772	1.551	327.1	286.4	6287	3094
2.40	5933	1.666	.128	1.794	1.574	330.8	290.2	6463	3588
2.60	5893	1.673	.138	1.811	1.591	331.7	291.4	6553	4083
2.80	5838	1.674	.147	1.821	1.601	330.5	290.5	6592	4465
3.00	5777	1.674	.151	1.826	1.605	327.8	288.2	6602	4637
3.20	5716	1.674	.153	1.827	1.607	324.6	285.5	6593	4692
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.58	5897	1.673	.137	1.810	1.589	331.7	291.3	6547	4045

PC= 800 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.662	.095	1.757	1.482	314.3	265.1	5496	2070
2.00	5885	1.673	.101	1.775	1.499	324.7	274.2	5976	2483
2.20	5939	1.688	.109	1.797	1.522	331.8	280.9	6287	2928
2.40	5933	1.702	.119	1.822	1.546	336.0	285.1	6463	3406
2.60	5893	1.712	.129	1.841	1.565	337.2	286.7	6553	3904
2.80	5838	1.714	.139	1.853	1.578	336.3	286.3	6592	4326
3.00	5777	1.715	.144	1.859	1.583	333.8	284.3	6602	4534
3.20	5716	1.715	.146	1.861	1.585	330.6	281.6	6593	4598
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.61	5889	1.713	.129	1.842	1.566	337.2	286.7	6558	3937

PC= 800 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.697	.087	1.784	1.417	319.1	253.4	5496	1921
2.00	5885	1.711	.092	1.803	1.435	329.7	262.5	5976	2308
2.20	5939	1.728	.099	1.827	1.459	337.2	269.4	6287	2727
2.40	5933	1.746	.108	1.854	1.487	341.9	274.1	6463	3185
2.60	5893	1.759	.118	1.877	1.509	343.8	276.5	6553	3675
2.80	5838	1.763	.129	1.892	1.525	343.3	276.7	6592	4136
3.00	5777	1.764	.135	1.899	1.532	341.0	275.1	6602	4398
3.20	5716	1.764	.137	1.902	1.534	337.9	272.6	6593	4481
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.66	5877	1.761	.121	1.882	1.515	343.9	276.8	6571	3826

TABLE III-20 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 800 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.741	.077	1.818	1.267	325.2	226.6	5496	1732
2.00	5885	1.756	.081	1.838	1.287	336.2	235.4	5976	2088
2.20	5939	1.777	.087	1.864	1.313	344.2	242.4	6287	2470
2.40	5933	1.800	.095	1.895	1.344	349.4	247.8	6463	2896
2.60	5893	1.817	.105	1.922	1.371	352.1	251.2	6553	3366
2.80	5838	1.826	.116	1.942	1.391	352.4	252.3	6592	3850
3.00	5777	1.828	.124	1.952	1.400	350.4	251.5	6602	4201
3.20	5716	1.829	.126	1.955	1.404	347.3	249.4	6593	4321
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.72	5860	1.823	.112	1.935	1.384	352.5	252.1	6580	3673

PC= 800 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.769	.072	1.841	1.106	329.3	197.9	5496	1613
2.00	5885	1.785	.075	1.860	1.126	340.3	205.9	5976	1947
2.20	5939	1.809	.080	1.889	1.154	348.7	213.0	6287	2307
2.40	5933	1.834	.087	1.921	1.186	354.3	218.8	6463	2709
2.60	5893	1.855	.097	1.951	1.217	357.4	222.8	6553	3159
2.80	5838	1.867	.107	1.974	1.239	358.2	224.8	6592	3643
3.00	5777	1.870	.118	1.986	1.251	356.6	224.6	6602	4052
3.20	5716	1.871	.119	1.990	1.255	353.5	223.0	6593	4208
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.76	5848	1.865	.105	1.970	1.236	358.2	224.6	6587	3558

PC= 800 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.788	.080	1.868	.949	334.1	169.8	5496	1535
2.00	5885	1.804	.080	1.884	.966	344.7	176.7	5976	1854
2.20	5939	1.830	.081	1.911	.993	352.8	183.3	6287	2191
2.40	5933	1.858	.085	1.943	1.024	358.3	188.9	6463	2575
2.60	5893	1.881	.092	1.973	1.055	361.4	193.2	6553	3007
2.80	5838	1.896	.101	1.997	1.078	362.4	195.7	6592	3484
3.00	5777	1.901	.110	2.011	1.092	361.1	196.2	6602	3927
3.20	5716	1.902	.114	2.016	1.098	358.2	195.0	6593	4121
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	5842	1.895	.100	1.995	1.077	362.4	195.6	6590	3449

TABLE III-21

LOX/RP-1, P_c = 900

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 900 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	.683	.561	1.244	1.228	222.6	219.6	5504	4992
2.00	5888	.673	.567	1.240	1.223	226.9	223.9	5991	5530
2.20	5936	.664	.572	1.236	1.220	228.1	225.1	6299	5900
2.40	5934	.658	.576	1.234	1.217	227.6	224.6	6484	6129
2.60	5897	.653	.580	1.232	1.216	225.9	222.9	6578	6252
2.80	5846	.654	.578	1.232	1.216	223.9	221.0	6625	6309
3.00	5786	.653	.579	1.232	1.216	221.6	218.6	6635	6327
3.20	5724	.653	.579	1.232	1.215	219.2	216.3	6627	6323
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.24	5940	.662	.573	1.236	1.219	228.2	225.1	6347	5959

PC= 900 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.231	.233	1.464	1.432	262.0	256.1	5504	3705
2.00	5888	1.224	.243	1.467	1.434	268.4	262.5	5991	4288
2.20	5936	1.215	.255	1.470	1.437	271.3	265.3	6299	4808
2.40	5934	1.209	.264	1.473	1.440	271.7	265.7	6484	5209
2.60	5897	1.205	.270	1.475	1.442	270.4	264.4	6578	5445
2.80	5846	1.203	.273	1.476	1.444	268.3	262.3	6625	5563
3.00	5786	1.202	.274	1.476	1.444	265.5	259.7	6635	5608
3.20	5724	1.202	.275	1.477	1.444	262.7	256.9	6627	5618
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.35	5938	1.210	.262	1.472	1.440	271.8	265.8	6445	5123

PC= 900 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.361	.186	1.547	1.498	276.7	267.9	5504	3267
2.00	5888	1.358	.195	1.552	1.504	284.1	275.2	5991	3825
2.20	5936	1.353	.207	1.561	1.512	288.0	279.0	6299	4365
2.40	5934	1.352	.204	1.556	1.507	287.0	278.0	6484	4831
2.60	5897	1.349	.211	1.560	1.511	285.9	276.9	6578	5138
2.80	5846	1.348	.214	1.562	1.513	283.9	275.0	6625	5294
3.00	5786	1.347	.216	1.563	1.514	281.1	272.3	6635	5356
3.20	5724	1.347	.217	1.563	1.514	278.2	269.4	6627	5373
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.26	5941	1.353	.208	1.561	1.512	288.2	279.2	6365	4507

PC= 900 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.435	.160	1.594	1.529	285.2	273.5	5504	2996
2.00	5888	1.434	.168	1.602	1.537	293.2	281.3	5991	3531
2.20	5936	1.434	.177	1.611	1.546	297.3	285.2	6299	4069
2.40	5934	1.432	.189	1.621	1.556	299.0	287.0	6484	4563
2.60	5897	1.430	.198	1.628	1.563	298.4	286.4	6578	4927
2.80	5846	1.429	.203	1.632	1.566	296.5	284.6	6625	5119
3.00	5786	1.428	.205	1.633	1.568	293.7	282.0	6635	5195
3.20	5724	1.428	.206	1.634	1.568	290.7	279.1	6627	5218
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.45	5929	1.432	.191	1.623	1.558	299.1	287.1	6514	4661

TABLE III-21 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 900 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.484	.144	1.628	1.546	291.2	276.6	5504	2806
2.00	5888	1.485	.152	1.637	1.556	299.7	284.7	5991	3322
2.20	5936	1.488	.163	1.650	1.569	304.5	289.5	6299	3851
2.40	5934	1.488	.174	1.663	1.581	306.7	291.7	6484	4357
2.60	5897	1.487	.184	1.671	1.590	306.4	291.4	6578	4762
2.80	5846	1.487	.188	1.675	1.593	304.4	289.6	6625	4987
3.00	5786	1.486	.191	1.677	1.595	301.5	286.9	6635	5077
3.20	5724	1.486	.192	1.678	1.596	298.5	284.0	6627	5106
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.47	5925	1.488	.178	1.666	1.585	306.9	291.9	6528	4514

PC= 900 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.519	.134	1.653	1.555	295.8	278.2	5504	2663
2.00	5888	1.523	.141	1.664	1.566	304.6	286.7	5991	3161
2.20	5936	1.527	.153	1.681	1.583	310.1	292.0	6299	3682
2.40	5934	1.531	.160	1.691	1.593	311.9	293.9	6484	4190
2.60	5897	1.531	.171	1.701	1.603	311.9	293.9	6578	4624
2.80	5846	1.530	.178	1.708	1.610	310.4	292.6	6625	4881
3.00	5786	1.530	.181	1.710	1.612	307.6	290.0	6635	4985
3.20	5724	1.530	.182	1.712	1.614	304.5	287.1	6627	5016
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.49	5921	1.532	.164	1.696	1.598	312.1	294.1	6539	4403

PC= 900 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.548	.125	1.673	1.559	299.3	278.9	5504	2549
2.00	5888	1.552	.133	1.685	1.571	308.5	287.6	5991	3033
2.20	5936	1.559	.143	1.702	1.588	314.1	293.0	6299	3542
2.40	5934	1.563	.155	1.718	1.604	316.9	295.9	6484	4052
2.60	5897	1.564	.166	1.730	1.616	317.2	296.2	6578	4506
2.80	5846	1.565	.171	1.736	1.621	315.4	294.6	6625	4790
3.00	5786	1.565	.173	1.738	1.624	312.6	292.0	6635	4907
3.20	5724	1.565	.175	1.739	1.625	309.4	289.1	6627	4944
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.52	5913	1.564	.162	1.726	1.612	317.3	296.3	6547	4350

PC= 900 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.570	.120	1.690	1.560	302.4	279.0	5504	2456
2.00	5888	1.576	.127	1.704	1.573	311.8	287.9	5991	2927
2.20	5936	1.585	.134	1.719	1.589	317.2	293.1	6299	3426
2.40	5934	1.591	.146	1.737	1.606	320.3	296.3	6484	3933
2.60	5897	1.593	.157	1.750	1.620	320.8	296.9	6578	4401
2.80	5846	1.593	.165	1.758	1.627	319.5	295.7	6625	4713
3.00	5786	1.593	.168	1.761	1.630	316.7	293.2	6635	4842
3.20	5724	1.593	.169	1.762	1.632	313.6	290.3	6627	4883
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.55	5907	1.593	.155	1.748	1.617	320.9	296.9	6560	4304

TABLE III-21 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 900 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.605	.110	1.716	1.553	307.0	277.8	5504	2309
2.00	5888	1.614	.116	1.730	1.566	316.6	286.7	5991	2758
2.20	5936	1.624	.125	1.749	1.586	322.8	292.7	6299	3241
2.40	5934	1.633	.136	1.769	1.606	326.3	296.2	6484	3740
2.60	5897	1.638	.147	1.785	1.621	327.2	297.2	6578	4224
2.80	5846	1.639	.156	1.794	1.631	326.0	296.4	6625	4580
3.00	5786	1.639	.159	1.798	1.635	323.3	294.0	6635	4734
3.20	5724	1.639	.161	1.799	1.636	320.2	291.1	6627	4784
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.59	5900	1.637	.146	1.784	1.621	327.2	297.2	6573	4193

PC= 900 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.632	.102	1.735	1.539	310.3	275.3	5504	2196
2.00	5888	1.642	.109	1.750	1.554	320.3	284.5	5991	2629
2.20	5936	1.654	.118	1.772	1.576	326.9	290.8	6299	3098
2.40	5934	1.665	.128	1.793	1.597	330.8	294.6	6484	3587
2.60	5897	1.672	.138	1.810	1.614	331.7	295.8	6578	4078
2.80	5846	1.674	.147	1.821	1.625	330.8	295.2	6625	4467
3.00	5786	1.674	.151	1.825	1.629	328.2	293.0	6635	4647
3.20	5724	1.674	.153	1.827	1.631	325.1	290.2	6627	4704
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.60	5896	1.672	.138	1.810	1.614	331.7	295.8	6579	4085

PC= 900 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.662	.095	1.757	1.512	314.4	270.6	5504	2070
2.00	5888	1.673	.101	1.774	1.529	324.7	279.9	5991	2482
2.20	5936	1.688	.110	1.797	1.552	331.7	286.5	6299	2932
2.40	5934	1.702	.119	1.821	1.576	335.9	290.7	6484	3407
2.60	5897	1.711	.128	1.839	1.594	337.2	292.3	6578	3899
2.80	5846	1.714	.138	1.853	1.608	336.6	292.1	6625	4325
3.00	5786	1.715	.143	1.858	1.613	334.2	290.1	6635	4541
3.20	5724	1.715	.145	1.860	1.615	331.0	287.4	6627	4610
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.64	5887	1.712	.130	1.843	1.598	337.2	292.4	6592	3992

PC= 900 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.697	.087	1.784	1.458	319.2	260.7	5504	1919
2.00	5888	1.710	.092	1.802	1.475	329.8	270.0	5991	2306
2.20	5936	1.728	.099	1.827	1.500	337.1	276.9	6299	2732
2.40	5934	1.745	.108	1.853	1.526	341.8	281.6	6484	3187
2.60	5897	1.757	.118	1.875	1.549	343.8	283.9	6578	3671
2.80	5846	1.763	.128	1.891	1.565	343.7	284.3	6625	4132
3.00	5786	1.763	.135	1.898	1.572	341.4	282.7	6635	4404
3.20	5724	1.764	.137	1.901	1.575	338.3	280.2	6627	4492
OPTIMUM MR CVS. IS VAC) POINT FOLLOWS.....									
2.69	5875	1.761	.122	1.883	1.557	344.0	284.3	6605	3882

TABLE III-21 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 900 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.741	.077	1.818	1.328	325.2	237.6	5504	1731
2.00	5888	1.756	.081	1.837	1.347	336.2	246.6	5991	2086
2.20	5936	1.777	.087	1.865	1.375	344.1	253.7	6299	2475
2.40	5934	1.799	.095	1.894	1.404	349.4	259.0	6484	2900
2.60	5897	1.816	.105	1.921	1.431	352.1	262.3	6578	3365
2.80	5846	1.826	.115	1.941	1.451	352.7	263.6	6625	3842
3.00	5786	1.827	.123	1.951	1.461	350.8	262.7	6635	4203
3.20	5724	1.828	.126	1.954	1.465	347.7	260.6	6627	4328
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.74	5861	1.824	.112	1.936	1.446	352.7	263.5	6616	3721

PC= 900 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.769	.072	1.841	1.188	329.3	212.5	5504	1612
2.00	5888	1.785	.075	1.860	1.207	340.4	220.9	5991	1946
2.20	5936	1.809	.080	1.889	1.236	348.6	228.1	6299	2312
2.40	5934	1.833	.088	1.921	1.267	354.3	233.8	6484	2713
2.60	5897	1.853	.096	1.950	1.297	357.4	237.7	6578	3159
2.80	5846	1.866	.107	1.973	1.320	358.5	239.8	6625	3633
3.00	5786	1.869	.115	1.985	1.331	356.9	239.4	6635	4050
3.20	5724	1.870	.119	1.989	1.336	353.9	237.7	6627	4215
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	5851	1.865	.106	1.971	1.318	358.5	239.7	6622	3590

PC= 900 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5755	1.788	.080	1.868	1.051	334.2	188.1	5504	1534
2.00	5888	1.805	.080	1.885	1.068	345.0	195.5	5991	1852
2.20	5936	1.831	.076	1.907	1.090	351.8	201.2	6299	2195
2.40	5934	1.857	.081	1.939	1.122	357.7	207.0	6484	2577
2.60	5897	1.880	.090	1.970	1.154	361.2	211.5	6578	3007
2.80	5846	1.895	.100	1.996	1.179	362.6	214.3	6625	3474
3.00	5786	1.900	.110	2.010	1.193	361.5	214.6	6635	3923
3.20	5724	1.901	.114	2.015	1.199	358.6	213.3	6627	4126
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.81	5843	1.896	.101	1.997	1.180	362.6	214.3	6626	3496

TABLE III-22

LOX/RP-1, P_c = 1,000

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 1000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	.684	.561	1.245	1.230	222.6	220.0	5508	4992
2.00	5889	.673	.567	1.240	1.225	227.0	224.3	6002	5537
2.20	5948	.664	.572	1.236	1.221	228.5	225.8	6328	5923
2.40	5946	.659	.575	1.234	1.219	228.1	225.4	6515	6157
2.60	5908	.656	.578	1.233	1.219	226.5	223.8	6611	6279
2.80	5854	.654	.579	1.233	1.218	224.3	221.6	6654	6335
3.00	5793	.653	.579	1.232	1.217	221.9	219.2	6665	6354
3.20	5732	.653	.579	1.232	1.217	219.5	216.8	6657	6351
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.25	5953	.662	.573	1.235	1.221	228.6	225.9	6393	6002

PC= 1000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.231	.233	1.464	1.435	261.9	256.7	5508	3703
2.00	5889	1.224	.243	1.467	1.437	268.5	263.1	6002	4285
2.20	5948	1.216	.254	1.470	1.440	271.7	266.3	6328	4814
2.40	5946	1.209	.264	1.473	1.444	272.3	266.8	6515	5223
2.60	5908	1.205	.270	1.475	1.446	270.9	265.5	6611	5464
2.80	5854	1.203	.273	1.476	1.447	268.6	263.2	6654	5580
3.00	5793	1.202	.274	1.477	1.447	265.9	260.6	6665	5627
3.20	5732	1.202	.275	1.477	1.447	263.1	257.9	6657	5638
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.36	5949	1.210	.262	1.473	1.443	272.3	266.9	6482	5148

PC= 1000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.361	.186	1.547	1.503	276.6	268.8	5508	3265
2.00	5889	1.358	.195	1.552	1.508	284.2	276.1	6002	3823
2.20	5948	1.354	.205	1.560	1.515	288.3	280.2	6328	4363
2.40	5946	1.353	.203	1.556	1.512	287.5	279.4	6515	4838
2.60	5908	1.350	.210	1.560	1.516	286.5	278.4	6611	5152
2.80	5854	1.348	.214	1.562	1.518	284.2	276.2	6654	5308
3.00	5793	1.347	.216	1.563	1.519	281.4	273.5	6665	5373
3.20	5732	1.347	.216	1.563	1.519	278.5	270.7	6657	5391
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.27	5954	1.353	.206	1.560	1.515	288.6	280.5	6406	4530

PC= 1000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.435	.160	1.594	1.536	285.2	274.7	5508	2993
2.00	5889	1.434	.168	1.602	1.543	293.2	282.5	6002	3529
2.20	5948	1.434	.178	1.612	1.553	298.0	287.1	6328	4064
2.40	5946	1.433	.188	1.621	1.562	299.6	288.8	6515	4565
2.60	5908	1.430	.198	1.628	1.569	299.0	288.2	6611	4938
2.80	5854	1.429	.202	1.631	1.573	296.8	286.2	6654	5131
3.00	5793	1.428	.205	1.633	1.574	294.0	283.4	6665	5211
3.20	5732	1.428	.205	1.633	1.575	291.0	280.6	6657	5235
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.44	5940	1.432	.190	1.623	1.564	299.7	288.8	6544	4658

TABLE III-22 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

L02/RP-1 12-27-57

PC= 1000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.484	.144	1.628	1.555	291.2	278.0	5508	2805
2.00	5889	1.485	.152	1.637	1.564	299.7	286.3	6002	3319
2.20	5948	1.488	.162	1.649	1.576	304.9	291.4	6328	3843
2.40	5946	1.489	.173	1.662	1.588	307.2	293.6	6515	4355
2.60	5908	1.488	.183	1.670	1.597	306.8	293.3	6611	4769
2.80	5854	1.486	.188	1.675	1.601	304.7	291.4	6654	4998
3.00	5793	1.486	.190	1.677	1.603	301.9	288.7	6665	5092
3.20	5732	1.486	.191	1.677	1.604	298.9	285.8	6657	5121
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.47	5936	1.489	.177	1.665	1.592	307.3	293.8	6559	4511

PC= 1000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.520	.134	1.653	1.565	295.7	280.0	5508	2662
2.00	5889	1.523	.141	1.664	1.576	304.6	288.5	6002	3159
2.20	5948	1.527	.151	1.678	1.590	310.2	293.9	6328	3672
2.40	5946	1.531	.162	1.693	1.604	312.8	296.5	6515	4185
2.60	5908	1.531	.172	1.703	1.615	312.8	296.6	6611	4629
2.80	5854	1.530	.178	1.708	1.620	310.8	294.7	6654	4889
3.00	5793	1.530	.180	1.710	1.622	308.0	292.1	6665	4998
3.20	5732	1.530	.181	1.711	1.623	304.9	289.2	6657	5032
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.50	5932	1.531	.167	1.698	1.610	313.1	296.9	6573	4411

PC= 1000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.548	.126	1.673	1.570	299.3	280.9	5508	2548
2.00	5889	1.552	.133	1.685	1.582	308.5	289.7	6002	3031
2.20	5948	1.558	.142	1.700	1.598	314.4	295.4	6328	3532
2.40	5946	1.564	.153	1.717	1.614	317.3	298.3	6515	4044
2.60	5908	1.565	.163	1.729	1.626	317.4	298.6	6611	4508
2.80	5854	1.565	.170	1.735	1.632	315.7	297.0	6654	4798
3.00	5793	1.565	.173	1.738	1.635	312.9	294.4	6665	4920
3.20	5732	1.564	.174	1.739	1.636	309.8	291.5	6657	4959
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.52	5926	1.565	.159	1.724	1.621	317.6	298.7	6577	4333

PC= 1000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.570	.120	1.690	1.573	302.3	281.3	5508	2456
2.00	5889	1.576	.127	1.703	1.586	311.8	290.3	6002	2925
2.20	5948	1.584	.134	1.718	1.601	317.7	296.0	6328	3415
2.40	5946	1.591	.145	1.736	1.619	320.9	299.1	6515	3924
2.60	5908	1.593	.157	1.750	1.632	321.4	299.8	6611	4401
2.80	5854	1.593	.164	1.757	1.640	319.8	298.4	6654	4719
3.00	5793	1.593	.168	1.761	1.643	317.0	295.9	6665	4854
3.20	5732	1.579	.176	1.755	1.637	312.7	291.7	6657	4914
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.55	5919	1.593	.154	1.747	1.629	321.5	299.8	6591	4293

TABLE III-22 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 1000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.606	.110	1.716	1.569	306.9	280.6	5508	2308
2.00	5889	1.614	.116	1.729	1.582	316.6	289.7	6002	2756
2.20	5948	1.623	.124	1.748	1.601	323.1	295.9	6328	3230
2.40	5946	1.633	.135	1.768	1.621	326.7	299.6	6515	3729
2.60	5908	1.638	.146	1.784	1.637	327.7	300.7	6611	4221
2.80	5854	1.638	.155	1.793	1.646	326.3	299.6	6654	4583
3.00	5793	1.638	.159	1.798	1.651	323.7	297.2	6665	4745
3.20	5732	1.623	.169	1.791	1.644	319.2	293.0	6657	4816
OPTIMUM MR CVS. IS VACD POINT FOLLOWS.....									
2.58	5912	1.637	.146	1.783	1.636	327.7	300.7	6605	4183

PC= 1000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.632	.102	1.735	1.558	310.2	278.7	5508	2195
2.00	5889	1.641	.109	1.750	1.574	320.3	288.1	6002	2627
2.20	5948	1.653	.117	1.770	1.593	327.2	294.6	6328	3086
2.40	5946	1.665	.127	1.792	1.616	331.2	298.6	6515	3575
2.60	5908	1.672	.138	1.810	1.634	332.4	300.0	6611	4072
2.80	5854	1.673	.146	1.820	1.643	331.1	299.1	6654	4468
3.00	5793	1.674	.151	1.825	1.648	328.6	296.8	6665	4656
3.20	5732	1.675	.152	1.827	1.650	325.5	294.1	6657	4715
OPTIMUM MR CVS. IS VACD POINT FOLLOWS.....									
2.60	5908	1.672	.138	1.810	1.634	332.4	300.0	6610	4069

PC= 1000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.662	.095	1.757	1.537	314.3	274.9	5508	2070
2.00	5889	1.673	.101	1.774	1.553	324.7	284.3	6002	2480
2.20	5948	1.687	.109	1.795	1.575	331.9	291.2	6328	2920
2.40	5946	1.701	.118	1.819	1.599	336.3	295.5	6515	3395
2.60	5908	1.711	.128	1.839	1.618	337.7	297.2	6611	3890
2.80	5854	1.714	.138	1.852	1.631	337.0	296.8	6654	4324
3.00	5793	1.714	.143	1.858	1.637	334.5	294.8	6665	4550
3.20	5732	1.715	.145	1.860	1.639	331.4	292.1	6657	4621
OPTIMUM MR CVS. IS VACD POINT FOLLOWS.....									
2.63	5900	1.712	.129	1.841	1.621	337.7	297.3	6622	3965

PC= 1000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.697	.087	1.784	1.490	319.0	266.4	5508	1922
2.00	5889	1.710	.092	1.801	1.507	329.7	275.9	6002	2306
2.20	5948	1.726	.099	1.825	1.531	337.4	283.0	6328	2721
2.40	5946	1.744	.108	1.851	1.557	342.2	287.8	6515	3173
2.60	5908	1.757	.117	1.874	1.580	344.2	290.2	6611	3659
2.80	5854	1.762	.128	1.890	1.596	344.0	290.5	6654	4127
3.00	5793	1.763	.134	1.898	1.604	341.7	288.8	6665	4409
3.20	5732	1.764	.137	1.901	1.607	338.6	286.3	6657	4501
OPTIMUM MR CVS. IS VACD POINT FOLLOWS.....									
2.68	5888	1.760	.121	1.881	1.587	344.4	290.6	6634	3847

TABLE III-22 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE C03 29SEPT, 60

LO2/RP-1 12-27-57

PC= 1000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.740	.077	1.817	1.376	325.0	246.1	5508	1736
2.00	5889	1.755	.081	1.837	1.396	336.2	255.5	6002	2086
2.20	5948	1.775	.087	1.862	1.421	344.3	262.8	6328	2466
2.40	5946	1.797	.095	1.892	1.451	349.7	268.2	6515	2887
2.60	5908	1.815	.104	1.920	1.479	352.5	271.6	6611	3350
2.80	5854	1.825	.115	1.939	1.499	352.9	272.7	6654	3835
3.00	5793	1.827	.123	1.950	1.509	351.1	271.8	6665	4205
3.20	5732	1.828	.126	1.954	1.513	348.1	269.6	6657	4337
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.73	5872	1.823	.111	1.934	1.493	353.0	272.5	6643	3689

PC= 1000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.768	.072	1.840	1.252	329.0	223.9	5508	1618
2.00	5889	1.784	.075	1.859	1.271	340.3	232.7	6002	1947
2.20	5948	1.806	.080	1.886	1.298	348.7	240.1	6328	2302
2.40	5946	1.831	.087	1.918	1.330	354.5	245.8	6515	2700
2.60	5908	1.853	.096	1.948	1.360	357.8	249.8	6611	3144
2.80	5854	1.865	.106	1.971	1.383	358.7	251.7	6654	3625
3.00	5793	1.869	.115	1.984	1.396	357.3	251.4	6665	4048
3.20	5732	1.870	.119	1.989	1.401	354.3	249.6	6657	4221
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.78	5860	1.864	.105	1.969	1.381	358.7	251.6	6651	3573

PC= 1000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5754	1.786	.080	1.866	1.131	333.7	202.3	5508	1540
2.00	5889	1.803	.080	1.883	1.148	344.7	210.2	6002	1857
2.20	5948	1.828	.081	1.909	1.174	353.0	217.1	6328	2192
2.40	5946	1.855	.085	1.940	1.205	358.6	222.8	6515	2569
2.60	5908	1.879	.091	1.970	1.235	361.8	226.8	6611	2993
2.80	5854	1.894	.100	1.994	1.259	362.8	229.1	6654	3464
3.00	5793	1.900	.109	2.009	1.274	361.8	229.5	6665	3917
3.20	5732	1.901	.114	2.015	1.280	358.9	228.0	6657	4132
OPTIMUM MR (VS. IS VAC) POINT FOLLOWS.....									
2.80	5853	1.894	.100	1.994	1.259	362.8	229.1	6654	3466

TABLE III-23

LOX/RP-1, P_c = 3,000

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV, 60

LO2/RP-1 SEPT.14,1960

PC= 3000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	.684	.561	1.245	1.240	222.8	221.9	5553	5011
2.20	5979	.669	.569	1.238	1.233	230.1	229.2	6505	6042
2.60	5973	.659	.575	1.233	1.229	229.0	228.1	6890	6513
3.00	5865	.655	.577	1.232	1.228	224.7	223.8	6973	6626
3.40	5743	.655	.577	1.232	1.227	219.9	219.1	6945	6608

PC= 3000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.232	.232	1.464	1.454	262.1	260.3	5553	3698
2.20	5979	1.219	.250	1.469	1.459	273.0	271.2	6505	4814
2.60	5973	1.207	.267	1.474	1.464	273.6	271.8	6890	5587
3.00	5865	1.204	.272	1.476	1.467	269.2	267.4	6973	5819
3.40	5743	1.203	.273	1.477	1.467	263.6	261.9	6945	5830

PC= 3000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.361	.186	1.547	1.532	276.8	274.2	5553	3262
2.20	5979	1.355	.203	1.558	1.543	289.5	286.8	6505	4344
2.60	5973	1.350	.207	1.557	1.543	289.1	286.4	6890	5226
3.00	5865	1.348	.214	1.561	1.547	284.7	282.0	6973	5538
3.40	5743	1.347	.215	1.562	1.547	278.9	276.3	6945	5562

PC= 3000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.434	.161	1.594	1.575	285.4	281.9	5553	2994
2.20	5979	1.434	.173	1.607	1.587	298.6	294.9	6505	4038
2.60	5973	1.432	.193	1.625	1.605	301.7	298.0	6890	4971
3.00	5865	1.428	.203	1.631	1.611	297.3	293.7	6973	5359
3.40	5743	1.428	.204	1.632	1.612	291.3	287.8	6945	5395

PC= 3000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.483	.143	1.626	1.602	291.0	286.7	5553	2804
2.20	5979	1.486	.159	1.645	1.620	305.7	301.2	6505	3817
2.60	5973	1.488	.179	1.667	1.643	309.5	305.0	6890	4776
3.00	5865	1.486	.190	1.675	1.651	305.5	301.0	6973	5225
3.40	5743	1.485	.191	1.677	1.652	299.4	295.0	6945	5271

PC= 3000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.518	.135	1.653	1.624	295.9	290.7	5553	2663
2.20	5979	1.525	.150	1.675	1.645	311.3	305.8	6505	3647
2.60	5973	1.531	.165	1.696	1.667	314.9	309.4	6890	4617
3.00	5865	1.530	.177	1.706	1.677	311.1	305.8	6973	5119
3.40	5743	1.530	.179	1.708	1.679	305.0	299.8	6945	5173

PC= 3000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.547	.125	1.672	1.637	299.2	293.1	5553	2548
2.20	5979	1.556	.140	1.696	1.661	315.1	308.8	6505	3507
2.60	5973	1.564	.160	1.724	1.690	320.1	313.7	6890	4482
3.00	5865	1.563	.173	1.736	1.702	316.5	310.2	6973	5034
3.40	5743	1.564	.175	1.739	1.704	310.4	304.3	6945	5096

TABLE III-23 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV. 60

LO2/RP-1 SEPT.14,1960

PC= 3000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.569	.120	1.689	1.650	302.4	295.4	5553	2455
2.20	5979	1.581	.131	1.712	1.673	318.2	311.0	6505	3391
2.60	5973	1.592	.151	1.743	1.704	323.6	316.3	6890	4365
3.00	5865	1.592	.165	1.757	1.718	320.3	313.2	6973	4958
3.40	5743	1.593	.168	1.760	1.721	314.3	307.3	6945	5029

PC= 3000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.604	.110	1.715	1.666	306.9	298.1	5553	2310
2.20	5979	1.619	.122	1.742	1.693	323.7	314.6	6505	3208
2.60	5973	1.635	.141	1.776	1.727	329.8	320.7	6890	4171
3.00	5865	1.637	.157	1.793	1.744	327.0	318.0	6973	4834
3.40	5743	1.638	.159	1.797	1.748	320.8	312.1	6945	4921

PC= 3000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.631	.102	1.733	1.674	310.2	299.7	5553	2198
2.20	5979	1.649	.115	1.764	1.705	327.8	316.9	6505	3065
2.60	5973	1.668	.133	1.801	1.743	334.4	323.5	6890	4015
3.00	5865	1.672	.148	1.820	1.761	331.9	321.2	6973	4731
3.40	5743	1.673	.151	1.824	1.765	325.6	315.1	6945	4835

PC= 3000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.661	.095	1.756	1.683	314.3	301.2	5553	2071
2.20	5979	1.682	.107	1.789	1.715	332.5	318.8	6505	2902
2.60	5973	1.706	.123	1.829	1.755	339.5	325.9	6890	3828
3.00	5865	1.713	.140	1.853	1.779	337.8	324.4	6973	4605
3.40	5743	1.713	.144	1.856	1.783	331.4	318.3	6945	4735

PC= 3000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.695	.087	1.782	1.684	319.0	301.5	5553	1928
2.20	5979	1.721	.097	1.818	1.720	337.9	319.7	6505	2704
2.60	5973	1.750	.114	1.864	1.766	346.1	327.9	6890	3598
3.00	5865	1.761	.131	1.892	1.794	344.9	327.1	6973	4441
3.40	5743	1.761	.135	1.897	1.799	338.6	321.1	6945	4609

PC= 3000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.738	.077	1.815	1.668	324.9	298.6	5553	1764
2.20	5979	1.769	.086	1.855	1.708	344.8	317.5	6505	2452
2.60	5973	1.806	.101	1.907	1.760	354.1	326.8	6890	3292
3.00	5865	1.823	.119	1.943	1.796	354.2	327.4	6973	4194
3.40	5743	1.825	.125	1.949	1.802	348.0	321.8	6945	4436

PC= 3000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.765	.071	1.837	1.641	328.7	293.7	5553	1677
2.20	5979	1.800	.079	1.879	1.683	349.1	312.7	6505	2291
2.60	5973	1.842	.092	1.935	1.739	359.2	322.8	6890	3089
3.00	5865	1.865	.111	1.976	1.780	360.3	324.5	6973	4004
3.40	5743	1.866	.117	1.984	1.788	354.1	319.2	6945	4317

TABLE III-23 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV, 60

LO2/RR-1 SEPT.14,1960

PC= 3000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5758	1.784	.069	1.853	1.608	331.6	287.8	5553	1625
2.20	5979	1.822	.074	1.896	1.651	352.4	306.9	6505	2173
2.60	5973	1.868	.086	1.954	1.709	362.8	317.4	6890	2941
3.00	5865	1.895	.105	2.000	1.755	364.7	320.0	6973	3851
3.40	5743	1.897	.113	2.010	1.765	358.8	315.1	6945	4225

TABLE III-24

LOX/RP-1, P_c = 5,000

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV. 60

LO2/RP-1 SEPT.14,1960

PC= 5000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	.686	.559	1.245	1.242	223.0	222.5	5573	5018
2.20	5993	.669	.569	1.239	1.236	230.7	230.2	6581	6095
2.60	6001	.661	.573	1.234	1.231	230.2	229.7	7019	6620
3.00	5899	.655	.578	1.233	1.230	226.1	225.5	7120	6757
3.40	5778	.654	.578	1.232	1.229	221.3	220.8	7093	6741

PC= 5000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.232	.232	1.464	1.458	262.2	261.1	5573	3697
2.20	5993	1.220	.248	1.468	1.462	273.5	272.4	6581	4814
2.60	6001	1.210	.264	1.475	1.469	275.1	274.0	7019	5632
3.00	5899	1.204	.271	1.476	1.470	270.6	269.5	7120	5910
3.40	5778	1.203	.273	1.476	1.470	265.1	264.0	7093	5926

PC= 5000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.361	.186	1.546	1.538	276.9	275.4	5573	3261
2.20	5993	1.355	.201	1.557	1.548	290.0	288.4	6581	4336
2.60	6001	1.353	.205	1.557	1.549	290.5	288.9	7019	5246
3.00	5899	1.348	.213	1.561	1.552	286.3	284.7	7120	5613
3.40	5778	1.348	.214	1.562	1.553	280.5	278.9	7093	5646

PC= 5000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.434	.160	1.594	1.582	285.5	283.4	5573	2993
2.20	5993	1.434	.171	1.605	1.593	299.0	296.8	6581	4027
2.60	6001	1.433	.191	1.623	1.611	302.8	300.6	7019	4979
3.00	5899	1.429	.201	1.631	1.619	299.0	296.9	7120	5423
3.40	5778	1.429	.203	1.632	1.620	293.1	291.0	7093	5469

PC= 5000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.483	.143	1.626	1.611	291.1	288.5	5573	2803
2.20	5993	1.486	.157	1.643	1.628	306.1	303.4	6581	3806
2.60	6001	1.488	.177	1.665	1.650	310.6	307.9	7019	4774
3.00	5899	1.486	.189	1.675	1.660	307.1	304.4	7120	5283
3.40	5778	1.485	.191	1.676	1.661	301.0	298.3	7093	5342

PC= 5000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.518	.135	1.653	1.635	296.0	292.8	5573	2662
2.20	5993	1.524	.149	1.673	1.655	311.6	308.3	6581	3636
2.60	6001	1.531	.163	1.694	1.676	315.9	312.6	7019	4606
3.00	5899	1.530	.176	1.705	1.688	312.7	309.5	7120	5170
3.40	5778	1.529	.178	1.707	1.689	306.6	303.4	7093	5241

PC= 5000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.547	.125	1.671	1.651	299.3	295.6	5573	2547
2.20	5993	1.555	.139	1.693	1.673	315.5	311.7	6581	3496
2.60	6001	1.563	.158	1.721	1.701	321.1	317.2	7019	4467
3.00	5899	1.563	.172	1.735	1.714	318.1	314.4	7120	5080
3.40	5778	1.563	.174	1.738	1.717	312.1	308.4	7093	5159

TABLE III-24 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV, 60

LO2/RR-1 SEPT. 14, 1960

PC= 5000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.569	.120	1.689	1.666	302.5	298.2	5573	2454
2.20	5993	1.580	.130	1.710	1.687	318.6	314.2	6581	3380
2.60	6001	1.591	.149	1.740	1.716	324.6	320.2	7019	4346
3.00	5899	1.592	.164	1.756	1.732	322.0	317.6	7120	5000
3.40	5778	1.592	.167	1.759	1.735	315.9	311.7	7093	5089

PC= 5000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.604	.110	1.714	1.685	307.0	301.7	5573	2309
2.20	5993	1.618	.121	1.739	1.710	324.0	318.6	6581	3198
2.60	6001	1.634	.139	1.773	1.744	330.7	325.2	7019	4148
3.00	5899	1.637	.155	1.792	1.763	328.6	323.2	7120	4866
3.40	5778	1.637	.158	1.795	1.766	322.4	317.2	7093	4977

PC= 5000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.631	.102	1.733	1.697	310.3	304.0	5573	2198
2.20	5993	1.647	.114	1.761	1.726	328.1	321.5	6581	3055
2.60	6001	1.666	.131	1.798	1.762	335.3	328.8	7019	3990
3.00	5899	1.672	.147	1.818	1.783	333.4	327.0	7120	4757
3.40	5778	1.672	.150	1.822	1.787	327.2	320.9	7093	4888

PC= 5000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.661	.095	1.756	1.711	314.4	306.5	5573	2074
2.20	5993	1.680	.107	1.786	1.742	332.7	324.5	6581	2893
2.60	6001	1.704	.121	1.825	1.781	340.4	332.1	7019	3801
3.00	5899	1.711	.139	1.849	1.805	339.1	331.0	7120	4625
3.40	5778	1.712	.143	1.854	1.810	333.1	325.1	7093	4782

PC= 5000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.695	.087	1.782	1.723	319.2	308.6	5573	1934
2.20	5993	1.718	.097	1.815	1.756	338.1	327.2	6581	2695
2.60	6001	1.747	.112	1.859	1.801	346.8	335.9	7019	3571
3.00	5899	1.760	.129	1.889	1.831	346.5	335.7	7120	4441
3.40	5778	1.761	.134	1.895	1.836	340.4	329.8	7093	4649

PC= 5000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.738	.078	1.815	1.727	325.1	309.3	5573	1785
2.20	5993	1.766	.086	1.852	1.764	345.0	328.6	6581	2444
2.60	6001	1.803	.099	1.902	1.813	354.7	338.3	7019	3267
3.00	5899	1.823	.117	1.940	1.851	355.7	339.5	7120	4170
3.40	5778	1.823	.124	1.947	1.858	349.6	333.8	7093	4470

PC= 5000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.765	.072	1.837	1.719	328.9	307.9	5573	1706
2.20	5993	1.797	.078	1.875	1.758	349.3	327.4	6581	2283
2.60	6001	1.838	.091	1.929	1.811	359.8	337.9	7019	3065
3.00	5899	1.863	.109	1.972	1.854	361.6	340.0	7120	3971
3.40	5778	1.864	.116	1.981	1.863	355.7	334.6	7093	4346

TABLE III-24 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV. 60

LO2/RP-1 SEPT.14,1960

PC= 5000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.783	.069	1.852	1.706	331.7	305.4	5573	1658
2.20	5993	1.819	.074	1.893	1.746	352.6	325.2	6581	2166
2.60	6001	1.863	.085	1.948	1.801	363.4	336.0	7019	2918
3.00	5899	1.892	.103	1.995	1.848	365.9	338.9	7120	3813
3.40	5778	1.895	.112	2.007	1.860	360.4	334.0	7093	4249

TABLE III-25

LOX/RP-1, P_c = 10,000

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV, 60

LO2/RR-1 SEPT.14,1960

PC=10000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	.685	.561	1.246	1.244	223.1	222.9	5595	5031
2.20	6009	.668	.572	1.239	1.238	231.5	231.2	6672	6158
2.60	6037	.662	.573	1.235	1.233	231.8	231.5	7189	6760
3.00	5945	.655	.577	1.233	1.232	227.9	227.6	7320	6933
3.40	5823	.656	.577	1.232	1.231	223.1	222.8	7297	6921

PC=10000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.233	.232	1.464	1.462	262.3	261.8	5595	3695
2.20	6009	1.221	.246	1.468	1.465	274.1	273.6	6672	4812
2.60	6037	1.211	.262	1.474	1.471	276.6	276.0	7189	5687
3.00	5945	1.206	.270	1.476	1.473	272.7	272.2	7320	6028
3.40	5823	1.204	.272	1.475	1.472	267.1	266.6	7297	6056

PC=10000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.361	.185	1.547	1.542	277.0	276.2	5595	3259
2.20	6009	1.356	.200	1.555	1.551	290.5	289.7	6672	4326
2.60	6037	1.353	.202	1.555	1.551	291.9	291.1	7189	5268
3.00	5945	1.349	.211	1.560	1.556	288.4	287.6	7320	5711
3.40	5823	1.348	.213	1.561	1.556	282.5	281.8	7297	5759

PC=10000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.434	.160	1.594	1.588	285.5	284.5	5595	2991
2.20	6009	1.434	.170	1.603	1.597	299.5	298.4	6672	4015
2.60	6037	1.433	.188	1.620	1.614	304.1	303.0	7189	4981
3.00	5945	1.429	.200	1.629	1.623	301.0	299.9	7320	5508
3.40	5823	1.428	.202	1.630	1.624	295.1	294.1	7297	5572

PC=10000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.483	.143	1.626	1.619	291.2	289.9	5595	2802
2.20	6009	1.485	.156	1.641	1.634	306.6	305.2	6672	3793
2.60	6037	1.488	.174	1.662	1.654	311.9	310.5	7189	4763
3.00	5945	1.486	.187	1.673	1.666	309.2	307.9	7320	5356
3.40	5823	1.485	.189	1.674	1.667	303.1	301.8	7297	5437

PC=10000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.518	.135	1.653	1.644	296.1	294.5	5595	2662
2.20	6009	1.523	.148	1.670	1.662	312.1	310.4	6672	3623
2.60	6037	1.530	.160	1.690	1.681	317.2	315.5	7189	4586
3.00	5945	1.530	.174	1.704	1.695	314.9	313.2	7320	5233
3.40	5823	1.529	.177	1.705	1.697	308.7	307.1	7297	5329

PC=10000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.547	.125	1.671	1.661	299.4	297.5	5595	2547
2.20	6009	1.554	.137	1.691	1.681	315.9	313.9	6672	3483
2.60	6037	1.563	.155	1.718	1.707	322.4	320.4	7189	4439
3.00	5945	1.563	.170	1.733	1.722	320.2	318.3	7320	5135
3.40	5823	1.563	.173	1.736	1.726	314.2	312.4	7297	5243

TABLE III-25 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV. 60

LO2/RP-1 SEPT.14,1960

PC=10000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.569	.120	1.689	1.677	302.5	300.4	5595	2456
2.20	6009	1.578	.129	1.708	1.696	319.0	316.8	6672	3367
2.60	6037	1.590	.146	1.736	1.724	325.8	323.6	7189	4315
3.00	5945	1.591	.162	1.753	1.741	324.0	321.9	7320	5047
3.40	5823	1.592	.165	1.757	1.745	318.1	315.9	7297	5168

PC=10000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.605	.110	1.715	1.700	307.1	304.5	5595	2310
2.20	6009	1.616	.120	1.737	1.722	324.4	321.7	6672	3186
2.60	6037	1.631	.137	1.768	1.753	331.8	329.0	7189	4116
3.00	5945	1.636	.153	1.789	1.774	330.7	327.9	7320	4900
3.40	5823	1.636	.157	1.793	1.778	324.6	321.9	7297	5050

PC=10000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.631	.102	1.733	1.716	310.5	307.3	5595	2201
2.20	6009	1.645	.113	1.758	1.741	328.4	325.1	6672	3044
2.60	6037	1.663	.129	1.792	1.775	336.4	333.1	7189	3957
3.00	5945	1.671	.144	1.815	1.798	335.5	332.2	7320	4779
3.40	5823	1.671	.149	1.819	1.802	329.3	326.1	7297	4956

PC=10000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.661	.095	1.756	1.734	314.5	310.5	5595	2085
2.20	6009	1.677	.106	1.783	1.761	333.1	328.9	6672	2882
2.60	6037	1.700	.121	1.820	1.798	341.6	337.5	7189	3769
3.00	5945	1.710	.136	1.847	1.824	341.3	337.2	7320	4628
3.40	5823	1.710	.141	1.852	1.830	335.2	331.2	7297	4844

PC=10000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.694	.088	1.782	1.753	319.2	313.9	5595	1961
2.20	6009	1.716	.096	1.812	1.782	338.4	332.9	6672	2685
2.60	6037	1.743	.110	1.853	1.823	347.7	342.2	7189	3537
3.00	5945	1.758	.127	1.885	1.855	348.3	342.9	7320	4429
3.40	5823	1.759	.133	1.892	1.862	342.5	337.1	7297	4702

PC=10000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.737	.079	1.816	1.772	325.2	317.3	5595	1828
2.20	6009	1.763	.085	1.848	1.804	345.3	337.0	6672	2435
2.60	6037	1.797	.097	1.894	1.850	355.5	347.3	7189	3235
3.00	5945	1.819	.114	1.933	1.889	357.3	349.2	7320	4135
3.40	5823	1.821	.122	1.943	1.899	351.7	343.7	7297	4512

PC=10000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.765	.074	1.838	1.779	329.2	318.7	5595	1754
2.20	6009	1.794	.078	1.871	1.812	349.6	338.6	6672	2275
2.60	6037	1.832	.089	1.921	1.862	360.6	349.5	7189	3035
3.00	5945	1.859	.106	1.965	1.906	363.2	352.3	7320	3924
3.40	5823	1.862	.115	1.977	1.918	357.8	347.2	7297	4380

TABLE III-25 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 14A 18 NOV, 60

LO2/RP-1 SEPT.14,1960

PC=10000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 1000010000

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.80	5761	1.783	.071	1.854	1.781	332.1	318.9	5595	1707
2.20	6009	1.815	.073	1.889	1.815	352.8	339.1	6672	2159
2.60	6037	1.857	.083	1.940	1.867	364.1	350.3	7189	2889
3.00	5945	1.888	.100	1.988	1.914	367.4	353.8	7320	3761
3.40	5823	1.892	.111	2.002	1.929	362.5	349.2	7297	4278

TABLE III-26

$N_2O_4/0.5N_2H_4 + 0.5UDMH$, $P_c = 50$

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

N204/.5N2H4+.5UDMH JAN 8,1960

PC= 50 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	.692	.555	1.247	.953	212.6	162.5	4466	4005
1.40	5678	.667	.569	1.237	.943	218.3	166.4	5206	4853
1.80	5631	.653	.578	1.232	.938	215.6	164.2	5462	5197
2.20	5481	.651	.580	1.231	.937	209.7	159.6	5475	5229
2.60	5322	.651	.580	1.231	.937	203.6	155.0	5404	5156

PC= 50 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.236	.224	1.461	.873	249.1	148.9	4466	2868
1.40	5678	1.221	.244	1.466	.878	258.7	155.0	5206	3796
1.80	5631	1.206	.268	1.474	.886	257.9	155.0	5462	4476
2.20	5481	1.203	.274	1.477	.889	251.6	151.5	5475	4616
2.60	5322	1.201	.274	1.475	.887	244.0	146.8	5404	4531

PC= 50 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.363	.177	1.540	.659	262.7	112.3	4466	2492
1.40	5678	1.356	.196	1.552	.670	274.0	118.3	5206	3373
1.80	5631	1.352	.206	1.558	.676	272.7	118.4	5462	4150
2.20	5481	1.347	.216	1.562	.681	266.2	116.0	5475	4389
2.60	5322	1.347	.215	1.561	.679	258.3	112.4	5404	4288

PC= 50 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.434	.152	1.586	.410	270.5	70.0	4466	2265
1.40	5678	1.434	.165	1.599	.423	282.2	74.7	5206	3101
1.80	5631	1.434	.190	1.624	.448	284.2	78.4	5462	3905
2.20	5481	1.427	.204	1.631	.455	277.9	77.6	5475	4239
2.60	5322	1.429	.202	1.631	.455	269.8	75.3	5404	4121

PC= 50 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.482	.134	1.616	.146	275.6	24.9	4466	2105
1.40	5678	1.485	.150	1.635	.166	288.7	29.3	5206	2906
1.80	5631	1.491	.174	1.665	.196	291.5	34.2	5462	3710
2.20	5481	1.487	.191	1.677	.208	285.7	35.4	5475	4120
2.60	5322	1.487	.188	1.675	.205	277.1	34.0	5404	3991

PC= 50 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.515	.126	1.641	.000	279.9	.0	4466	1989
1.40	5678	1.522	.141	1.664	.000	293.6	.0	5206	2758
1.80	5631	1.534	.159	1.693	.000	296.4	.0	5462	3552
2.20	5481	1.532	.177	1.709	.000	291.2	.0	5475	4024
2.60	5322	1.532	.174	1.706	.000	282.2	.0	5404	3882

PC= 50 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.542	.116	1.659	.000	282.9	.0	4466	1894
1.40	5678	1.553	.131	1.683	.000	297.1	.0	5206	2636
1.80	5631	1.567	.153	1.720	.000	301.1	.0	5462	3419
2.20	5481	1.566	.173	1.739	.000	296.2	.0	5475	3945
2.60	5322	1.565	.170	1.735	.000	287.0	.0	5404	3791

TABLE III-26 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N204/.5N2H4+.5UDMH JAN 8,1960

PC= 50 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.564	.111	1.675	.000	285.8	.0	4466	1818
1.40	5678	1.576	.125	1.701	.000	300.3	.0	5206	2537
1.80	5631	1.595	.144	1.738	.000	304.3	.0	5462	3306
2.20	5481	1.595	.165	1.760	.000	299.8	.0	5475	3876
2.60	5322	1.594	.161	1.755	.000	290.4	.0	5404	3707

PC= 50 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.597	.102	1.699	.000	289.8	.0	4466	1699
1.40	5678	1.613	.114	1.728	.000	304.9	.0	5206	2380
1.80	5631	1.637	.133	1.770	.000	309.8	.0	5462	3124
2.20	5481	1.640	.155	1.796	.000	305.9	.0	5475	3759
2.60	5322	1.640	.151	1.791	.000	296.3	.0	5404	3562

PC= 50 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.622	.096	1.718	.000	293.0	.0	4466	1611
1.40	5678	1.641	.106	1.747	.000	308.3	.0	5206	2258
1.80	5631	1.669	.125	1.794	.000	314.0	.0	5462	2980
2.20	5481	1.675	.146	1.822	.000	310.4	.0	5475	3661
2.60	5322	1.676	.141	1.817	.000	300.5	.0	5404	3440

PC= 50 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.650	.087	1.737	.000	296.3	.0	4466	1511
1.40	5678	1.672	.098	1.770	.000	312.4	.0	5206	2119
1.80	5631	1.705	.116	1.821	.000	318.7	.0	5462	2813
2.20	5481	1.715	.138	1.853	.000	315.7	.0	5475	3537
2.60	5322	1.716	.131	1.847	.000	305.5	.0	5404	3286

PC= 50 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.682	.080	1.761	.000	300.4	.0	4466	1399
1.40	5678	1.708	.089	1.797	.000	317.1	.0	5206	1957
1.80	5631	1.748	.104	1.852	.000	324.1	.0	5462	2608
2.20	5481	1.765	.127	1.892	.000	322.3	.0	5475	3365
2.60	5322	1.763	.120	1.883	.000	311.5	.0	5404	3085

PC= 50 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.721	.071	1.792	.000	305.6	.0	4466	1279
1.40	5678	1.753	.077	1.830	.000	323.0	.0	5206	1749
1.80	5631	1.800	.091	1.891	.000	331.0	.0	5462	2345
2.20	5481	1.828	.113	1.941	.000	330.7	.0	5475	3094
2.60	5322	1.823	.106	1.929	.000	319.1	.0	5404	2803

PC= 50 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.745	.066	1.811	.000	308.8	.0	4466	1221
1.40	5678	1.782	.070	1.851	.000	326.8	.0	5206	1612
1.80	5631	1.833	.083	1.916	.000	335.4	.0	5462	2173
2.20	5481	1.868	.103	1.972	.000	335.9	.0	5475	2894
2.60	5322	1.862	.096	1.958	.000	323.9	.0	5404	2609

TABLE III-26 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

N2O4/.5N2H4+.5UDMH JAN 8,1960

PC= 50 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5487	1.762	.063	1.825	.000	311.4	.0	4466	1185
1.40	5678	1.801	.065	1.866	.000	329.4	.0	5206	1520
1.80	5631	1.857	.077	1.934	.000	338.6	.0	5462	2047
2.20	5481	1.897	.096	1.994	.000	339.7	.0	5475	2741
2.60	5322	1.889	.089	1.979	.000	327.3	.0	5404	2466

TABLE III-27

$N_2O_4/0.5N_2H_4$ = 0.5UDMH, P_c = 300

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N204/.5N2H4+.5UDMH JAN 8,1960

PC= 300 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	.695	.554	1.249	1.200	213.0	204.6	4513	4018
1.40	5715	.676	.564	1.240	1.191	220.3	211.6	5402	4966
1.80	5722	.659	.575	1.234	1.185	219.5	210.8	5781	5454
2.20	5582	.655	.578	1.232	1.183	213.8	205.3	5815	5521
2.60	5413	.655	.577	1.232	1.183	207.4	199.2	5720	5424

PC= 300 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.238	.223	1.462	1.364	249.4	232.6	4513	2864
1.40	5715	1.226	.239	1.464	1.367	260.2	242.8	5402	3775
1.80	5722	1.212	.260	1.472	1.374	261.7	244.3	5781	4539
2.20	5582	1.206	.270	1.476	1.378	256.2	239.2	5815	4795
2.60	5413	1.206	.269	1.475	1.377	248.2	231.7	5720	4676

PC= 300 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.364	.177	1.541	1.394	262.9	237.8	4513	2489
1.40	5715	1.358	.191	1.549	1.402	275.2	249.1	5402	3342
1.80	5722	1.353	.211	1.564	1.417	278.2	252.0	5781	4133
2.20	5582	1.350	.211	1.561	1.414	270.9	245.4	5815	4523
2.60	5413	1.350	.209	1.559	1.412	262.4	237.7	5720	4382

PC= 300 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.435	.152	1.587	1.391	270.7	237.3	4513	2261
1.40	5715	1.434	.161	1.595	1.399	283.3	248.5	5402	3068
1.80	5722	1.435	.180	1.615	1.419	287.2	252.4	5781	3854
2.20	5582	1.431	.198	1.629	1.433	282.7	248.7	5815	4340
2.60	5413	1.431	.195	1.626	1.431	273.7	240.7	5720	4179

PC= 300 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.483	.134	1.616	1.371	275.7	233.9	4513	2102
1.40	5715	1.484	.146	1.630	1.385	289.6	246.1	5402	2874
1.80	5722	1.489	.165	1.655	1.410	294.3	250.7	5781	3642
2.20	5582	1.487	.185	1.672	1.427	290.2	247.7	5815	4203
2.60	5413	1.488	.181	1.670	1.425	281.0	239.7	5720	4019

PC= 300 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.516	.126	1.642	1.348	280.1	229.9	4513	1987
1.40	5715	1.520	.138	1.658	1.364	294.6	242.3	5402	2728
1.80	5722	1.531	.151	1.681	1.388	299.1	246.8	5781	3476
2.20	5582	1.531	.172	1.702	1.409	295.4	244.4	5815	4089
2.60	5413	1.532	.167	1.699	1.405	285.9	236.5	5720	3886

PC= 300 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.543	.116	1.659	1.316	283.0	224.5	4513	1892
1.40	5715	1.549	.128	1.677	1.334	297.9	237.0	5402	2607
1.80	5722	1.562	.145	1.707	1.364	303.6	242.6	5781	3341
2.20	5582	1.565	.167	1.732	1.389	300.6	241.1	5815	3991
2.60	5413	1.565	.162	1.727	1.385	290.7	233.0	5720	3773

TABLE III-27 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

N2O4/.5N2H4+.5UDMH JAN 8,1960

PC= 300 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.564	.111	1.676	1.284	285.9	219.0	4513	1816
1.40	5715	1.573	.122	1.695	1.303	301.1	231.5	5402	2510
1.80	5722	1.589	.136	1.724	1.332	306.7	237.0	5781	3226
2.20	5582	1.594	.159	1.753	1.361	304.1	236.1	5815	3906
2.60	5413	1.594	.153	1.747	1.355	294.0	228.0	5720	3674

PC= 300 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.598	.102	1.700	1.210	289.9	206.4	4513	1699
1.40	5715	1.609	.112	1.721	1.231	305.7	218.6	5402	2354
1.80	5722	1.629	.126	1.755	1.265	312.1	224.9	5781	3045
2.20	5582	1.638	.149	1.787	1.297	310.1	225.1	5815	3764
2.60	5413	1.637	.143	1.780	1.290	299.6	217.2	5720	3507

PC= 300 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.622	.096	1.718	1.130	293.1	192.8	4513	1615
1.40	5715	1.636	.103	1.740	1.152	309.0	204.6	5402	2234
1.80	5722	1.659	.118	1.777	1.189	316.1	211.5	5781	2903
2.20	5582	1.672	.140	1.812	1.224	314.4	212.4	5815	3642
2.60	5413	1.671	.135	1.805	1.218	303.8	204.9	5720	3370

PC= 300 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.650	.088	1.738	1.003	296.5	171.1	4513	1527
1.40	5715	1.667	.096	1.762	1.028	313.1	182.5	5402	2097
1.80	5722	1.693	.110	1.803	1.068	320.7	190.0	5781	2739
2.20	5582	1.712	.130	1.842	1.108	319.7	192.2	5815	3485
2.60	5413	1.710	.123	1.833	1.098	308.5	184.8	5720	3202

PC= 300 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.681	.081	1.762	.782	300.6	133.5	4513	1444
1.40	5715	1.702	.087	1.788	.809	317.7	143.7	5402	1936
1.80	5722	1.733	.099	1.832	.853	325.9	151.7	5781	2540
2.20	5582	1.759	.119	1.878	.899	325.9	155.9	5815	3275
2.60	5413	1.755	.114	1.868	.889	314.4	149.5	5720	2991

PC= 300 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.720	.074	1.794	.324	306.0	55.3	4513	1356
1.40	5715	1.745	.076	1.821	.352	323.5	62.4	5402	1731
1.80	5722	1.783	.086	1.870	.400	332.5	71.2	5781	2283
2.20	5582	1.819	.105	1.924	.455	333.9	78.9	5815	2977
2.60	5413	1.812	.099	1.911	.441	321.6	74.3	5720	2708

PC= 300 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.745	.069	1.814	.000	309.5	.0	4513	1305
1.40	5715	1.774	.068	1.842	.000	327.2	.0	5402	1596
1.80	5722	1.815	.079	1.894	.000	336.8	.0	5781	2115
2.20	5582	1.857	.096	1.953	.000	338.9	.0	5815	2774
2.60	5413	1.848	.090	1.938	.000	326.1	.0	5720	2517

TABLE III-27 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N204/.5N2H4+.5UDMH JAN 8,1960

PC= 300 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.763	.066	1.829	.000	312.0	.0	4513	1271
1.40	5715	1.793	.064	1.857	.000	329.8	.0	5402	1505
1.80	5722	1.837	.074	1.911	.000	339.9	.0	5781	1992
2.20	5582	1.884	.089	1.973	.000	342.4	.0	5815	2625
2.60	5413	1.874	.084	1.957	.000	329.3	.0	5720	2376

TABLE III-28

$N_2O_4/0.5N_2H_4 = 0.5UDMH$, $P_c = 500$

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

M204/.5UDMH+.5N2H4 FEB 22,1960

PC= 500 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	.695	.554	1.249	1.219	213.0	208.0	4521	4021
1.40	5721	.677	.564	1.241	1.212	220.7	215.5	5446	4990
1.80	5744	.661	.574	1.235	1.205	220.4	215.2	5867	5521
2.20	5610	.655	.578	1.233	1.203	215.0	209.8	5911	5604
2.60	5437	.656	.577	1.233	1.204	208.4	203.4	5808	5498
3.00	5270	.658	.576	1.234	1.204	202.1	197.3	5656	5335
5.00	4583	.674	.566	1.240	1.210	176.6	172.4	4784	4394
7.00	4079	.687	.558	1.245	1.216	157.9	154.2	4011	3608
9.00	3708	.695	.554	1.248	1.219	143.9	140.5	3424	3043

PC= 500 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.238	.223	1.462	1.403	249.4	239.4	4521	2863
1.40	5721	1.227	.237	1.464	1.406	260.4	250.0	5446	3770
1.80	5744	1.214	.257	1.471	1.412	262.7	252.2	5867	4545
2.20	5610	1.206	.269	1.475	1.417	257.3	247.0	5911	4842
2.60	5437	1.208	.267	1.475	1.416	249.3	239.4	5808	4710
3.00	5270	1.210	.263	1.473	1.414	241.3	231.7	5656	4485
5.00	4583	1.224	.241	1.465	1.407	208.8	200.4	4784	3360
7.00	4079	1.233	.229	1.463	1.404	185.5	178.0	4011	2631
9.00	3708	1.239	.222	1.461	1.403	168.5	161.7	3424	2160

PC= 500 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.364	.177	1.541	1.453	262.9	247.9	4521	2488
1.40	5721	1.358	.190	1.548	1.460	275.4	259.7	5446	3335
1.80	5744	1.353	.209	1.562	1.474	278.9	263.2	5867	4123
2.20	5610	1.350	.210	1.560	1.472	272.0	256.6	5911	4558
2.60	5437	1.351	.208	1.558	1.470	263.4	248.5	5808	4402
3.00	5270	1.353	.202	1.555	1.467	254.8	240.4	5656	4138
5.00	4583	1.358	.193	1.551	1.463	220.9	208.4	4784	2981
7.00	4079	1.362	.182	1.544	1.456	195.8	184.6	4011	2299
9.00	3708	1.365	.175	1.540	1.452	177.5	167.4	3424	1870

PC= 500 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.435	.152	1.587	1.469	270.7	250.7	4521	2260
1.40	5721	1.434	.160	1.594	1.476	283.5	262.6	5446	3061
1.80	5744	1.435	.178	1.612	1.495	287.9	266.9	5867	3837
2.20	5610	1.431	.197	1.628	1.510	283.8	263.3	5911	4366
2.60	5437	1.431	.194	1.625	1.507	274.6	254.8	5808	4188
3.00	5270	1.433	.187	1.620	1.502	265.4	246.1	5656	3895
5.00	4583	1.435	.163	1.597	1.480	227.5	210.8	4784	2739
7.00	4079	1.435	.156	1.591	1.473	201.7	186.8	4011	2092
9.00	3708	1.436	.150	1.585	1.468	182.8	169.2	3424	1689

PC= 500 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.482	.134	1.616	1.469	275.7	250.7	4521	2102
1.40	5721	1.484	.146	1.629	1.483	289.8	263.7	5446	2868
1.80	5744	1.489	.163	1.652	1.505	294.9	268.7	5867	3624
2.20	5610	1.487	.184	1.671	1.524	291.4	265.7	5911	4220
2.60	5437	1.488	.180	1.668	1.521	281.9	257.1	5808	4019
3.00	5270	1.489	.172	1.661	1.514	272.1	248.0	5656	3708
5.00	4583	1.485	.148	1.633	1.486	232.7	211.7	4784	2566
7.00	4079	1.484	.137	1.621	1.474	205.6	187.0	4011	1945
9.00	3708	1.484	.131	1.615	1.468	186.1	169.2	3424	1561

TABLE III-28 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

N2O4/.5UDMH+.5N2H4 FEB 22,1960

PC= 500 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.516	.126	1.641	1.465	280.1	250.0	4521	1987
1.40	5721	1.520	.137	1.657	1.481	294.7	263.4	5446	2722
1.80	5744	1.528	.154	1.682	1.506	300.3	268.8	5867	3458
2.20	5610	1.531	.170	1.701	1.525	296.6	265.9	5911	4099
2.60	5437	1.532	.165	1.697	1.521	286.8	257.0	5808	3880
3.00	5270	1.531	.158	1.689	1.512	276.7	247.8	5656	3557
5.00	4583	1.522	.140	1.661	1.485	236.7	211.5	4784	2434
7.00	4079	1.518	.129	1.647	1.471	208.9	186.5	4011	1834
9.00	3708	1.517	.122	1.640	1.463	189.0	168.7	3424	1465

PC= 500 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.543	.116	1.659	1.453	283.0	247.9	4521	1893
1.40	5721	1.549	.127	1.676	1.470	298.1	261.5	5446	2602
1.80	5744	1.560	.143	1.703	1.498	304.2	267.4	5867	3322
2.20	5610	1.565	.165	1.730	1.524	301.7	265.8	5911	3997
2.60	5437	1.565	.160	1.725	1.519	291.6	256.8	5808	3763
3.00	5270	1.563	.152	1.716	1.510	281.1	247.4	5656	3432
5.00	4583	1.551	.129	1.680	1.475	239.4	210.1	4784	2324
7.00	4079	1.546	.118	1.665	1.459	211.1	185.0	4011	1743
9.00	3708	1.544	.114	1.658	1.452	191.2	167.4	3424	1387

PC= 500 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.564	.111	1.676	1.440	285.9	245.8	4521	1817
1.40	5721	1.572	.122	1.694	1.459	301.2	259.4	5446	2504
1.80	5744	1.586	.134	1.720	1.485	307.2	265.2	5867	3207
2.20	5610	1.593	.157	1.750	1.515	305.2	264.2	5911	3907
2.60	5437	1.593	.151	1.744	1.509	294.8	255.0	5808	3660
3.00	5270	1.591	.143	1.734	1.499	284.0	245.5	5656	3324
5.00	4583	1.575	.123	1.698	1.463	242.0	208.5	4784	2234
7.00	4079	1.568	.113	1.682	1.447	213.3	183.4	4011	1668
9.00	3708	1.566	.107	1.672	1.437	192.8	165.7	3424	1323

PC= 500 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.597	.102	1.699	1.405	290.0	239.8	4521	1703
1.40	5721	1.608	.112	1.719	1.426	305.8	253.5	5446	2349
1.80	5744	1.626	.125	1.750	1.457	312.6	260.1	5867	3027
2.20	5610	1.637	.147	1.784	1.490	311.1	259.9	5911	3756
2.60	5437	1.636	.141	1.777	1.483	300.4	250.7	5808	3488
3.00	5270	1.632	.133	1.765	1.471	289.2	241.1	5656	3149
5.00	4583	1.612	.113	1.724	1.430	245.6	203.8	4784	2091
7.00	4079	1.603	.103	1.706	1.412	216.3	179.1	4011	1548
9.00	3708	1.599	.096	1.695	1.401	195.4	161.5	3424	1222

PC= 500 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.622	.096	1.718	1.365	293.1	232.9	4521	1625
1.40	5721	1.635	.103	1.738	1.386	309.2	246.4	5446	2229
1.80	5744	1.656	.117	1.773	1.420	316.6	253.6	5867	2886
2.20	5610	1.671	.138	1.809	1.456	315.5	254.0	5911	3627
2.60	5437	1.669	.133	1.802	1.449	304.6	245.0	5808	3348
3.00	5270	1.664	.125	1.789	1.436	293.1	235.3	5656	3010
5.00	4583	1.639	.104	1.743	1.390	248.3	198.1	4784	1979
7.00	4079	1.628	.096	1.724	1.371	218.6	173.9	4011	1459
9.00	3708	1.623	.089	1.712	1.360	197.4	156.7	3424	1145

TABLE III-28 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N204/.9UDMH+.5N2H4 FEB 22,1960

PC= 500 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.649	.088	1.738	1.297	296.5	221.3	4521	1544
1.40	5721	1.666	.095	1.761	1.320	313.2	234.8	5446	2092
1.80	5744	1.690	.108	1.798	1.357	321.1	242.4	5867	2723
2.20	5610	1.711	.128	1.839	1.398	320.6	243.8	5911	3464
2.60	5437	1.708	.122	1.829	1.388	309.2	234.7	5808	3179
3.00	5270	1.700	.116	1.816	1.375	297.5	225.3	5656	2846
5.00	4583	1.670	.095	1.766	1.325	251.6	188.8	4784	1848
7.00	4079	1.657	.086	1.743	1.303	221.1	165.2	4011	1352
9.00	3708	1.650	.081	1.731	1.290	199.6	148.7	3424	1056

PC= 500 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.680	.082	1.762	1.175	300.7	200.4	4521	1468
1.40	5721	1.700	.087	1.787	1.199	317.8	213.3	5446	1932
1.80	5744	1.730	.098	1.827	1.240	326.3	221.3	5867	2524
2.20	5610	1.757	.117	1.874	1.286	326.8	224.3	5911	3246
2.60	5437	1.752	.112	1.864	1.276	315.0	215.7	5808	2967
3.00	5270	1.743	.105	1.847	1.259	302.6	206.3	5656	2642
5.00	4583	1.706	.086	1.792	1.204	255.3	171.5	4784	1694
7.00	4079	1.691	.077	1.767	1.180	224.1	149.6	4011	1224
9.00	3708	1.681	.071	1.753	1.165	202.1	134.3	3424	950

PC= 500 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.719	.075	1.794	.912	306.1	155.6	4521	1384
1.40	5721	1.744	.075	1.820	.938	323.6	166.8	5446	1727
1.80	5744	1.779	.086	1.865	.983	333.0	175.5	5867	2269
2.20	5610	1.816	.103	1.919	1.038	334.7	180.9	5911	2946
2.60	5437	1.808	.098	1.906	1.024	322.1	173.1	5808	2684
3.00	5270	1.795	.091	1.886	1.005	309.1	164.6	5656	2376
5.00	4583	1.751	.073	1.824	.943	259.9	134.3	4784	1489
7.00	4079	1.731	.065	1.796	.915	227.8	116.0	4011	1063
9.00	3708	1.719	.060	1.779	.898	205.1	103.5	3424	817

PC= 500 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.745	.070	1.815	.639	309.7	109.1	4521	1333
1.40	5721	1.773	.068	1.840	.665	327.3	118.2	5446	1592
1.80	5744	1.810	.078	1.888	.712	337.1	127.2	5867	2102
2.20	5610	1.853	.094	1.948	.772	339.6	134.6	5911	2745
2.60	5437	1.844	.089	1.933	.757	326.7	128.0	5808	2495
3.00	5270	1.829	.083	1.912	.736	313.2	120.6	5656	2200
5.00	4583	1.779	.065	1.844	.668	262.7	95.2	4784	1357
7.00	4079	1.756	.058	1.814	.638	230.1	81.0	4011	960
9.00	3708	1.742	.053	1.796	.620	207.0	71.5	3424	733

PC= 500 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5489	1.763	.067	1.831	.361	312.3	61.6	4521	1298
1.40	5721	1.792	.064	1.855	.386	330.0	68.6	5446	1499
1.80	5744	1.832	.073	1.905	.435	340.2	77.7	5867	1980
2.20	5610	1.880	.087	1.968	.498	343.1	86.8	5911	2596
2.60	5437	1.869	.082	1.951	.482	329.8	81.4	5808	2355
3.00	5270	1.853	.077	1.930	.460	316.2	75.4	5656	2069
5.00	4583	1.798	.060	1.858	.389	264.8	55.4	4784	1261
7.00	4079	1.773	.053	1.826	.357	231.6	45.2	4011	885
9.00	3708	1.756	.049	1.807	.338	208.3	38.9	3424	674

TABLE III-29

$N_2O_4/0.5N_2\vec{H}_4 = 0.5UDMH,\ P_c = 1,000$

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N2O4/.5N2H4+.5UDMH JAN 8,1960

PC= 1000 AE/AT= 1.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	.696	.553	1.249	1.234	213.1	210.6	4529	4022
1.40	5728	.680	.563	1.242	1.228	221.2	218.6	5498	5015
1.80	5771	.662	.573	1.236	1.221	221.7	219.0	5979	5605
2.20	5645	.656	.577	1.233	1.219	216.4	213.8	6039	5712
2.60	5469	.657	.577	1.233	1.219	209.7	207.2	5925	5595

PC= 1000 AE/AT= 2.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.239	.223	1.462	1.433	249.4	244.4	4529	2862
1.40	5728	1.228	.236	1.464	1.435	260.7	255.5	5498	3764
1.80	5771	1.216	.254	1.470	1.441	263.7	258.4	5979	4549
2.20	5645	1.205	.268	1.474	1.444	258.6	253.4	6039	4905
2.60	5469	1.208	.266	1.474	1.445	250.6	245.6	5925	4754

PC= 1000 AE/AT= 3.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.365	.176	1.541	1.497	262.9	255.4	4529	2487
1.40	5728	1.359	.189	1.548	1.504	275.7	267.8	5498	3328
1.80	5771	1.354	.206	1.560	1.516	279.9	271.9	5979	4109
2.20	5645	1.349	.209	1.558	1.514	273.3	265.6	6039	4601
2.60	5469	1.352	.205	1.557	1.513	264.7	257.2	5925	4423

PC= 1000 AE/AT= 4.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.435	.152	1.587	1.528	270.7	260.7	4529	2260
1.40	5728	1.433	.163	1.596	1.538	284.3	273.8	5498	3056
1.80	5771	1.435	.175	1.609	1.551	288.7	278.2	5979	3816
2.20	5645	1.431	.195	1.626	1.567	285.3	275.0	6039	4393
2.60	5469	1.432	.191	1.623	1.564	275.9	265.9	5925	4193

PC= 1000 AE/AT= 5.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.483	.134	1.617	1.543	275.8	263.2	4529	2101
1.40	5728	1.484	.145	1.629	1.555	290.0	276.9	5498	2862
1.80	5771	1.488	.160	1.648	1.575	295.7	282.5	5979	3602
2.20	5645	1.487	.182	1.669	1.595	292.9	280.0	6039	4237
2.60	5469	1.488	.177	1.665	1.592	283.1	270.6	5925	4015

PC= 1000 AE/AT= 6.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.517	.126	1.642	1.554	280.2	265.1	4529	1985
1.40	5728	1.519	.137	1.656	1.568	294.9	279.2	5498	2716
1.80	5771	1.527	.151	1.678	1.590	301.1	285.2	5979	3436
2.20	5645	1.531	.168	1.699	1.610	298.1	282.6	6039	4108
2.60	5469	1.531	.162	1.694	1.606	287.9	272.9	5925	3869

PC= 1000 AE/AT= 7.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.544	.116	1.660	1.557	283.1	265.6	4529	1893
1.40	5728	1.549	.126	1.675	1.572	298.3	279.9	5498	2596
1.80	5771	1.558	.141	1.699	1.596	304.8	286.4	5979	3299
2.20	5645	1.564	.163	1.727	1.624	303.1	285.1	6039	3999
2.60	5469	1.564	.157	1.721	1.618	292.6	275.1	5925	3746

TABLE III-29 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV. 60

N204/.5N2H4+.5UDMH JAN 8,1960

PC= 1000 AE/AT= 8.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.565	.111	1.676	1.559	286.0	265.9	4529	1821
1.40	5728	1.571	.121	1.693	1.575	301.4	280.5	5498	2499
1.80	5771	1.584	.132	1.716	1.598	307.9	286.8	5979	3186
2.20	5645	1.593	.154	1.747	1.629	306.5	285.9	6039	3902
2.60	5469	1.592	.148	1.740	1.622	295.8	275.8	5925	3639

PC= 1000 AE/AT= 10.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.597	.103	1.699	1.552	289.9	264.8	4529	1719
1.40	5728	1.607	.111	1.718	1.571	306.0	279.8	5498	2344
1.80	5771	1.623	.123	1.746	1.599	313.2	286.8	5979	3006
2.20	5645	1.636	.144	1.781	1.634	312.5	286.7	6039	3739
2.60	5469	1.635	.138	1.773	1.626	301.4	276.4	5925	3463

PC= 1000 AE/AT= 12.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.621	.097	1.718	1.542	293.1	263.0	4529	1650
1.40	5728	1.634	.103	1.737	1.561	309.3	277.9	5498	2224
1.80	5771	1.653	.115	1.768	1.591	317.1	285.5	5979	2866
2.20	5645	1.670	.136	1.806	1.630	316.9	286.0	6039	3602
2.60	5469	1.667	.130	1.797	1.621	305.5	275.5	5925	3320

PC= 1000 AE/AT= 15.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.648	.090	1.738	1.518	296.5	258.9	4529	1578
1.40	5728	1.665	.095	1.760	1.539	313.3	274.1	5498	2088
1.80	5771	1.686	.107	1.793	1.572	321.6	282.1	5979	2704
2.20	5645	1.708	.125	1.834	1.613	321.8	283.1	6039	3430
2.60	5469	1.704	.121	1.825	1.605	310.3	272.8	5925	3150

PC= 1000 AE/AT= 20.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.680	.084	1.764	1.470	300.9	250.8	4529	1507
1.40	5728	1.699	.086	1.786	1.492	317.9	265.6	5498	1928
1.80	5771	1.725	.096	1.821	1.528	326.8	274.0	5979	2507
2.20	5645	1.754	.115	1.869	1.575	328.1	276.5	6039	3207
2.60	5469	1.748	.110	1.858	1.564	315.8	265.9	5925	2938

PC= 1000 AE/AT= 30.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.720	.076	1.797	1.356	306.5	231.3	4529	1424
1.40	5728	1.744	.075	1.818	1.378	323.8	245.3	5498	1721
1.80	5771	1.774	.085	1.858	1.417	333.4	254.3	5979	2253
2.20	5645	1.812	.101	1.913	1.472	335.7	258.3	6039	2908
2.60	5469	1.803	.096	1.899	1.458	322.8	247.9	5925	2656

PC= 1000 AE/AT= 40.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.746	.072	1.817	1.229	310.0	209.7	4529	1373
1.40	5728	1.771	.068	1.839	1.251	327.4	222.7	5498	1590
1.80	5771	1.804	.077	1.881	1.293	337.5	232.0	5979	2087
2.20	5645	1.848	.092	1.940	1.353	340.5	237.3	6039	2708
2.60	5469	1.838	.088	1.926	1.338	327.4	227.4	5925	2468

TABLE III-29 (cont.)

ENGINE DESIGN HANDBOOK DATA. TAPE 12 18 NOV, 60

N2O4/.5N2H4+.5UDMH JAN 8,1960

PC= 1000 AE/AT= 50.00 ALTITUDE=SEA LEVEL ID= 2000025003

MR	C*	V/C*	PA/PCAT	CFVAC	CFALT	ISVAC	ISALT	TC	T
1.00	5488	1.765	.069	1.833	1.099	312.8	187.4	4529	1336
1.40	5728	1.790	.063	1.854	1.119	330.1	199.2	5498	1498
1.80	5771	1.826	.072	1.898	1.163	340.5	208.7	5979	1966
2.20	5645	1.874	.085	1.960	1.225	343.9	215.0	6039	2561
2.60	5469	1.863	.081	1.944	1.209	330.4	205.5	5925	2328